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Hamilton Standard

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UH-1H AIDAPS TEST BED PROGRAM
VOLUME I OF II

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U.S. ARMY AVIATION SYSTEMS COMMAND
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13. ABSTRACT

The UH-1H helicopter Test Bed Program was accomplished at the US Army Aeronautical Depot Maintenance Center, (ARADMAC), Corpus Christi, Texas, during the period 4 October 1970 through 17 December 1971. The program objective was to determine the feasibility of state-of-the-art hardware to automatically accomplish inspection, diagnostic and prognostic maintenance functions on selected subsystems of the UH-1H helicopter. The Hamilton Standard hardware for the program is identified as Airborne Integrated Diagnostic System (AIDS). Helicopter components, both serviceable and degraded, were run and monitored for malfunction discrimination by the AIDS in ARADMAC test cells and in two UH-1H aircraft. Trending for prognosis was attempted while accumulating flight time on two additional UH-1H aircraft utilizing serviceable components. The test results demonstrated the objectives of the Test Bed Program.

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Digital Data Software						
Digital Data Acquisition						
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— SUBMITTED TO —

THE U.S. ARMY AVIATION SYSTEMS COMMAND

ST. LOUIS, MISSOURI

— IN ACCORDANCE WITH —

CONTRACT NO. DAA J01–70–C–0827 (P3L)

UH-1H AIDAPS TEST BED PROGRAM

Hamilton Standard
DIVISION OF UNITED AIRCRAFT CORPORATION
WINDSOR LOCKS CONNECTICUT · USA

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ABSTRACT

The UH-1H helicopter test bed program was accomplished at the US Army Aeronautical Depot Maintenance Center (ARADMAC), Corpus Christi, Texas, during the period 4 October 1970 through 17 December 1971. The program objective was to determine the feasibility of state-of-the-art hardware to automatically accomplish inspection, diagnostic and prognostic maintenance functions on selected subsystems of the UH-1H helicopter. The Hamilton Standard hardware for the program is identified as Airborne Integrated Diagnostic System. Helicopter components, both serviceable and degraded, were run and monitored for malfunction discrimination by the AIDS in ARADMAC test cells and in two UH-1H aircraft. Trending for prognosis was attempted while accumulating flight time on two additional UH-1H aircraft utilizing serviceable components. The test results demonatrated the objectives of the test bed program.

FOREWORD

The UH-1 Helicopter Test Bed Program was conducted for the US Army Aviation Systems Command under a contract (No. DAAJ01-70-C-0827(P3L) with Hamilton Standard Division of United Aircraft Corporation. This program is a subelement of the Department of the Army RD&E project (1F164204DC3201) to develop an Automatic Inspection, Diagnostic and Prognostic System (AIDAPS) for Army aircraft. The overall program is in response to a Qualitative Materiel Requirement for an AIDAPS which was approved by DA in October 1967.

GOVERNMENT ASSESSMENT OF PHASE E VERIFICATION TEST

Section 10 of this report documents the accomplishments under Phase E of the program. Phase E was a test of the Accuracy and Repeatability of the Hamilton equipment. Page 10-18 of Volume I summarizes the Hamilton Standard diagnosis of the helicopter components (both serviceable or good and degraded or bad) which were implanted by the Government in the UH-1H aircraft.

The table below lists the actual conditions of the test components implanted in the UH-1H aircraft monitored by Hamilton Standard:

Conditions	Date	Engine	Tranmission	90° Gear Box	42° Gear Box
1	19 Nov 71	Bad	Bad	Good	Bad
2	24 Nov 71	Bad	Bad	Good	Bad*
3	3 Dec 71	Good	Bad	Bad	Bad
4	7 Dec 71	Good	Bad	Bad	Bad
5	9 Dec 71	Good	Good	Bad	Bad
6	10 Dec 71	Good	Good	Bad	Bad
7	14 Dec 71	Good	Good	Bad	Good
8	16 Dec 71	Good	Good	Bad*	Good

*The component conditions noted with an asterisk (above) were revealed to the contractor prior to his final analysis and are not included in the percentage scores shown below. The remaining component conditions had not previously been identified to the contractor.

An overall diagnostic accuracy of 90% was obtained by Hamilton Standard in determining the conditions (Bad or Good) of the implanted engines, transmissions, 90° gear boxes and 42° gear boxes.

Although the Hamilton Standard equipment did achieve objectives of the Test Bed Program to demonstrate state-of-the-art capability, the foregoing results are not satisfactory for immediate hardware implementation.

The efforts expended by the Hamilton Standard Corporation and assigned personnel were very commendable. The above efforts and the knowledge accumulated will be used in subsequent steps during development of AIDAPS.

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SECTION 1.0

INTRODUCTION

1.0 INTRODUCTION

1.1 Report Summary

This document fully details all activities carried out by the Hamilton Standard Division of United Aircraft under the Automatic Inspection, Diagnostics, and Prognostics (AIDAPS) UH-1H Test Bed Program contract DAAJ01-70-C-0827 (P3L) issued by the U.S. Army Aviation Systems Command. Contract performance commenced in July 1970 and will extend through March 1972.

The report is divided into ten main sections contained in two volumes. Volume I contains all of the text material. Volume II includes all of the figures followed by the tables divided up into sections so arranged that they accompany the text for that section.

Section 1 discusses the program philosophy and goals. Section 2 highlights the program conduct and achievements. Section 3 explains the program tasks performed and their purpose. Section 4 describes the hardware employed by Hamilton Standard. Sections 5, 6 and 7 are the technical heart of the report. These sections completely explain the diagnostic techniques utilized and document the test results. Section 7 presents the Phase D Verification Test Analysis. Sections 8 and 9 summarize the conclusions and recommendations, respectively, which are drawn from the detailed text performance descriptions for all phases of the program. Section 10 is considered as an Appendix covering Phase E work and results.

1.2 Discussion of Program Philosophy

The U.S. Army has a requirements to evaluate automated inspection, diagnostic, and prognostic systems leading to a fully functional unit that achieves increased aircraft availability and lower maintenance costs by:

1. Reducing unwarranted removals, and
2. Reducing fault isolation maintenance time.

To meet the above goals, Hamilton Standard has worked with AVSCOM through the Test Bed Program to generate the necessary software to perform diagnostics and prognostics on major subsystems of the UH-1H vehicle. The use of a Hamilton Standard airborne system with on-board real-time processing allowed for a comprehensive evaluation of total system capabilities to fulfill Army requirements.

The approach in the Test Bed Program was to collect data on pertinent parameters of the UH-1H subsystems a) while the subsystem is operating nominally, and b) while known degraded parts have been implanted within the subsystem. The above two groups of data are then compared and differences noted for generation of the proper fault isolation message.

By implanting known degraded parts, much more pertinent data can be generated and fault isolation software accomplished during a given period of time. However, implementation of this approach engenders such problems as:

1. The degraded parts chosen are not really functional degraded, or so slight so as not to cause differences from normal signatures.
2. During the build-up of the subsystem with the degraded parts, the signature changes because of the build-up.

1.2 Continued

3. The signature pattern generated is not a true response because the "housing" the degraded part was placed into test was not the same one that the part became faulty in. This effectively places the part in an artifical environment and thus causes differences from the real environment.

It was anticipated that in the course of the program the above jeopardies would be minor risks when compared to the benefit of the controlled experiment with an accelerated time schedule.

1.3 Hamilton Standard Effectively Met All Goals

The purposes of the Test Bed Program were:

1. To develop logic that will isolate malfunctions on the UH-1H aircraft.
2. To determine the capability of existing state-of-the-art hardware.
3. To collect sufficient data to be used as a data base.
4. To perform malfunction detection and verification tests on a UH-1H aircraft.
5. To predict through trending techniques the remaining time before component removal is required.

Hamilton Standard, in its performance during the program, has effectively met all the goals.

The use of the modified system developed by Hamilton Standard for the KSS (KLM Royal Dutch Airlines, Scandinavian Airlines System and Swissair) group of airlines on their DC-9 aircraft provided readily available existing hardware for the program.

The following software logic was developed, proved out or implemented:

1. Gas Path Analysis
2. Lycoming Logic Package
3. Other Mechanical Logic (Hydraulic system, electrical)
4. Vibration

The Gas Path Analysis developed from Hamilton's extensive fuel control experience was shown able to detect the implanted degraded thermodynamic parts. The Vibration Analysis concept and results are

1.3 Continued

very promising in that a logic approach has been used which has detected the degraded parts with a high degree of confidence. In performing Phases B, D and E, a large amount of properly documented data was taken. This has assured a sufficient data base. Malfunction detection and verification was successfully performed in Phases D and E. Trending was accomplished, but the accumulated flight time was not high enough to confirm the remaining life predictions.

SECTION 2.0

PROGRAM CONDUCT AND ACHIEVEMENT HIGHLIGHTS

2.0 PROGRAM CONDUCT AND ACHIEVEMENT HIGHLIGHTS

2.1 Program Conduct

The Test Bed Program was structured to achieve the following objectives: (1) automation of inspection methods, (2) fault isolation via diagnostics, and (3) life remaining predictions via prognostication. The program scope included use and evaluation of state-of-the-art equipment, non-interference with test vehicle subsystems, use of existing sensors to maximum extent possible, development of data base for this and interim programs, and determination of AIDAPS potential as a maintenance tool.

The program plan involved four basic phases plus the addition of Phase E. Phase A allowed three months for the preparation and delivery of two "off-the-shelf" state-of-the-art hardware systems. Test cell baseline malfunction signature data was gathered and analyzed during the 2 month Phase B period. One month was provided as Phase C to install the AIDAPS sensors and hardware in two UH-1H helicopters at ARADMAC. Flight testing of AIDAPS with known good and bad parts occupied 6 months of Phase D. Two more months were added to Phase D by verification flight tests with unknown bad parts implanted. Phase E included a two month period for rotor monitoring and worse degraded parts flight tests plus transmission testing to failure in a test cell. The program phases are summarized in Figure 2-1.

2.2 Program Highlights

2.2.1 Phase B

- 178 test runs were achieved instead of the planned 142. 28 engines, 63 transmissions, 52 90° gearboxes, and 35 42° gearboxes were instrumented and analyzed.
- Signature analysis verified ability of diagnostic techniques to discriminate between good and bad parts.
- AIDAPS hardware and instrumentation successfully performance tested.
- 100% detection of bearing and gear faults by vibration analysis technique.
- 80-90% detection of engine compressor and turbine section faults by Gas Path Analysis.
- Existing Gas Path Analysis method successfully applied to T53-L13 engine.
- Extensive test cell data base obtained.
- Test cell environment enabled detection of minor component failures having very low detection threshold.

2.2.2 Phase C

- 2 UH-1H's instrumented as indicated in Figure 2-2.
- AIDAPS installed and tested for no interference with aircraft systems.
- AIDAPS given flight safety approval.

2.2.3 Phase D Flight Tests

- Provided flight refinement of software limits and data recording criteria.

- Obtained trending data for prognostication. Initial trend data (245 hours) within norms expected.

- Verified test cell gas path analysis conclusions.

- Indicated refinements necessary in vibration analysis to account for transmissibility, aerodynamic noise, and mounting. Baseline mean data scatter was higher than in test cell runs.

- 74-81% malfunction detection (6 borderline cases) in 80 vibration test cases with known bad implants including engines, transmissions, 42° and 90° gearboxes.

- "Degree of badness" of vibration causing implants generally low.

- Validity of diagnostic methods to detect marginal parts indicates capability to track part deterioration while in service.

2.2.4 Phase D Verification Test

- Six sets of unknown implanted bad parts test flown.

- On July 23, 1971, AIC 61011 was flown with unknown bad engine compressor, bad 42° gearbox input roller, good 90° gearbox, and good transmission. All degraded LRU's were correctly diagnosed and no good LRU's were judged faulty.

- Overall diagnostic fault isolation score calculated by Hamilton Standard to be 88%.

2.2.5 Phase E

- Rotor out of track and out of balance diagnostics found feasible for integration within AIDAPS.

- Data from first transmission test to failure indicates vibration analysis would have predicted functional failure 40 hours before occurence.

- Worse degraded parts testing gave more positive measure of AIDAPS diagnostic effectivity. Eight flights accomplished instead of 6 projected.

2.2.6 Overall Highlights

- Over 4000 vibration narrow band spectrum analyses were performed on more than 1,264,000 data inputs.

- About 60 rolls of magnetic tape were utilized with three-quarters of a mile of data on each.

- Approximately 5 miles of computer printouts were generated and analyzed.

- Demonstrated AIDAPS feasibility for helicopters through Hamilton Standard's airborne digital processor approach.

- Developed and verified software techniques such as limit analysis, vibration, and gas path application required for AIDAPS implementation now.

2.3 Hardware Utilized

Hamilton Standard chose an off-the-shelf airborne digital processor, Airborne Integrated Data System (AIDS), for application in the UH-1H Test Bed Program. The on-board system configuration chosen has the following characteristics:

1. Automatic data scanning and signal conversion
2. Easy modification of diagnostic limits
3. Automatic data compression through flight mode recognition and digital recording of airborne data.
4. Flexibility and ease of modification through program software.

The Hamilton Standard airborne AIDAPS processor has a fully programmable magnetic core memory and a high speed 16-bit parallel arithmetic unit which permits timely and complete system flexibility. From an operational standpoint, the system's airborne operation was fully automatic and did not require any flight crew attention. The use of such items as a Flight Data Entry Panel allowed for instant read-out of parameters to check data validity and system operation before the test flights. Via ground memory loader, the system operation was altered in a manner of minutes from a prepared punched tape. Due to the tight schedule and quantity of parameters used, the choice of an airborne digital processor was particularly advantageous in meeting the objectives of a trial program because the expected changes could be implemented readily through software modification rather than hardware revision. The hardware included an airborne go/no go Maintenance Action Annunciator Panel (MAAP) to display in real time aircraft system replace or adjustment messages generated by the processor output.

2.4 Developed Diagnostic Software

The diagnostic software used in the UH-1H Test Bed Program can be split into four main headings:

* Gas Path
* Vibration
* Mechanical/Electrical Limit Checking
* Trending

The gas path logic was generated by Hamilton Standard many years ago to fill the need within the gas turbine industry for an analytical approach to predicting the impossible to measure parameters that effect engine health such as efficiency changes, areas changes, compressor pumping capacity changes, and turbine inlet temperature changes. Early in the program, the generalized equations were fitted to the Lycoming T53-L13 engine and incorporated within the airborne system.

The development of the vibration software, as illustrated in Figure 2-3, was more emperical in nature. It became apparent early in the evolution of the Hamilton Standard vibration approach that the problem of applying vibration analysis to a helicopter propulsion system had never before been treated in depth or successfully achieved. Much original creative work was carried out during the Test Bed Program which for the first time brought helicopter power train system health analysis through vibration to the point of practical feasibility.

The steps involved in arriving at the Hamilton Standard vibration analysis technique are outlined in Figure 2-3 and fully explained in

2.4 (Continued)

Sections 6.3 through 6.5. Hamilton Standard feels that this work represents some of the most significant results from the Test Bed Program.

Mechanical/electrical limit checking software was implemented based upon discrete parameter monitoring to perform diagnostics on the engine accessory systems, oil lubrication system, hydraulic system, and electrical system. The logic was devised based upon the Army supplied data package and information furnished by Lycoming.

Techniques applied by Hamilton Standard in implementing the logic included fixed, adaptive, and floating limits based upon parameter operational characteristics. Multi-parameter cross-correlation to achieve a higher degree of fault isolation and verification was also employed. Parameter limits were established and verified. Most flight data was within the exceedance limits since the systems monitored were not deliberately rendered operationally faulty.

Trend software was devised to perform health prognostication on the monitored parameters. The time history of the individual parameter was extrapolated to predict the life remaining until exceedance of a diagnostic limit. A large number of these plots and predictions are included in Section 5. The initial trend data for the 245 hours of flight testing just overhauled systems was within the norms as would be expected.

2.5 Data Processing Performed and System Outputs

The data processing performed can be considered in two parts:

1. On-board processing

2. Off-site processing

2.5.1 On-Board Processing

The airborne hardware used has the full capability to process the data required to perform mechanical and gas path diagnostics. The system output went to two places. The first was to a digital recorder for use in an off-site processor such as an IBM 370 for refinement of the software; the second was to the Maintenance Action Annunciator Panel (MAAP) which displayed the diagnostic messages with electro/mechanical indicators on board the helicopter.

2.5.2 Off-Site Processing

The airborne recorded digital tapes were returned to Hamilton Standard after a review of the tapes was made on the on-site DDP-116 processor for completeness of data. These tapes were then processed at Hamilton Standard to confirm and improve the on-board software. The data was also used for the trending (prognostication) software. Output of the processing was standard IBM hard copy print-out.

2.5.3 Vibration Data Analysis

The vibration data was airborne recorded using temporary analog equipment. This analog data was then digitized to allow the use of an off-site processor to perform the manipulations required. Comparison summary sheets which compared each transducer's output for a particular flight to the generated mean of the good flights were generated as the output of the vibration data analysis.

SECTION 3

THE PROGRAM

3.0 THE PROGRAM

3.1 Program Tasks and Organization

Section 3 explains how the Test Bed Program was conducted to fulfill the goals outlined in Section 1.2. These goals can be considered in terms of their inspection, diagnostic, and prognostic impact on Army airmobile operations. The relation between furthering these goals AIDAPS and the structure and operation of the Test Bed Program is initially expanded upon in Sections 3.2 - 3.5. The various program tasks carried out to achieve and verify the goals are then detailed for each of the Phases A through D. Phase E is treated separately as an Appendix in Section 10. The Test Bed Program was strongly goal oriented, and this emphasis on results and evaluation set the program theme.

3.2 Inspection Impact

The Hamilton Standard AIDAPS system was designed to significantly improve the efficiency with which maintenance personnel routinely accomplish aircraft sy.tems inspections after the aircraft has landed or prior to takeoff. Present turnaround procedures are based upon performing a visual inspection or the use of limited data from the flight crew.

Inspection advantages are achieved through (1) automation of inspection procedures; (2) a substantial reduction in inspection times; and (3) decreasing required personnel skill levels.

3.2.1 Automation of Inspection Procedures

From an operational standpoint, the UH-1 AIDAPS system is fully automatic and does not require any flight crew attention. Also, since it is an airborne system, the mechanical conditi'n of the aircraft is constantly being monitored during flight; i.e., the aircraft inspection is completed during the flight and diagnostic messages will be outputted in a concise manner and displayed on a Maintenance Action Annunciator Panel. The information is thus available to the ground crew immediately after landing.

Another important aspect is that each parameter is being continuously monitored once every two seconds for proper operation. Normally, ground crew inspection is limited to discrete preflight inspection and post flight checks. These inspections cannot practically check many parameters which are comprehensively included within the AIDAPS airborne system.

3.2.2 A Substantial Reduction in Inspection Times

Since the AIDSAPS system is a completely automatic airborne system, inspection is continuously accomplished during the flight of the aircraft.

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3.2.2 A Substantial Reduction in Inspection Times

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3.2.2 Continued

Many routine ground inspection tasks are eliminated with attendant time savings. Because the various aircraft systems are monitored during flight, the changes in parameters analyzed, diagnostics performed, and messages read out on the Maintenance Action Annunciator Panel, the aircraft inspection health status is immediately available to ground personnel. This method of airborne inspection eliminates the need for an extensive postflight inspection. Postflight inspection can be confined to inspection of aircraft structure, other visual damage, etc., that cannot be monitored utilizing the AIDAPS airborne system. In some cases, the AIDAPS airborne system may not fully diagnose a subtle mechanical problem, but can help the ground personnel pinpoint the problem by selecting an area for further troubleshooting.

The airborne AIDAPS system, in addition to reducing inspection time, is more thorough than manual inspections; i.e., many more parameters are monitored than is possible by manually reading the available aircraft instrumentation. Therefore, a more thorough preflight, during flight, and post flight inspection is accomplished using the airborne AIDAPS system. An inherent advantage of the AIDAPS system in reducing inspection time is that the difficult task of duplicating flight problems on the ground is no longer necessary since the AIDAPS system is constantly gathering data during the flight.

3.2.3 <u>Decreasing Required Personnel Skill Level</u>

Various types of inspections are performed on aircraft by assorted maintenance personnel of different skill levels. For example, daily preflight and postflight inspections are normally performed as indicated by the crew chief. These checks include checking fluid levels, plugged filters, foreign object damage, etc. The AIDAPS system will perform most of these inspections and permit the crew chief to perform

3.2.3 Continued

troubleshooting functions that were normally performed by specialists.

The use of the airborne AIDAPS system tends to reduce the required skill level of maintenance personnel engaged in line maintenance activities. Each crew chief, by utilizing the AIDAPS systems, can effectively troubleshoot the various aircraft systems without outside assistance. The use of specialists for line maintenance activity can, therefore, be reduced to the troubleshooting of more difficult and subtle maintenance problems that are beyond the scope of the AIDAPS system.

Also, periodic inspections and intermediate inspections are normally performed by qualified mechanics. These inspections include visual external inspection of engines, borescope inspection of internal engine sections, hot section inspections, calibration of some sensors, inspection of accessory systems, etc. Using the AIDAPS system, these inspections are quickly and accurately performed by mechanics with a lower skill level than would be required if they were done manually.

Various different configurations of automatic inspection aids and displays can be incorporated into the AIDAPS system to evaluate and display selected parameters so as to most effectively help maintenance personnel do their job. Typical of the maintenance interfaces available are:

1. Visual indicators readily visible from ground level.
2. Information recorded and displayed on-board whenever aircraft electrical power is actuated.

3.2.3 Continued

3. Information recorded and stored on-board and displayed by ground support equipment.

4. Inclusion with visual indicators of an integral engine-mounted "use and abuse" history recorder which manually displays historical data.

3.3 Diagnostic Impact

The Hamilton Standard AIDAPS system is designed to have a direct and immediate benefit to all forward operational organizational level units. This benefit is derived from increased operational availability of the aircraft, decreased requirements for maintenance diagnostic skills and time, and a decrease in spare parts usage. The Hamilton Standard AIDAPS system provides to the using organization capabilities that do not now exist or that require extensive skill development and time to acquire. This is especially relevant due to rapid turnover of personnel in the military. At the same time, the system will furnish the operational commander a real-time status report for each aircraft and increase the operational availability within his present resources.

3.3.1 Increased Aircraft Availability

Aircraft availability is primarily determined by the maintenance time to: (a) conduct periodic maintenance inspections, and (b) trouble shoot, isolate, repair, and test to insure correction of failures by unscheduled maintenance.

Probably the most significant improvement attributed to an airborne diagnostic system is the increased availability obtained by reducing the time required for both the above maintenance actions in data compression and immediate on-the-spot diagnostics. The reduction of maintenance man hour per flying hour associated with the reduced time for problem identification and components isolation; added trouble shooting capability of each crew chief;

3.3.1 Continued

and reduction in scope of periodic maintenance adds up to a 15-20% improvement in availability of operational aircraft considering only the maintenance action aspect. Improved spares level through reduction in false removals and improved data base for overhaul and repair will further contribute to the improved availability.

3.3.2 Decreased Maintenance Time

Crew chief experience will in many cases pinpoint a failure and identify corrective action based on repeated cases. However, if an airborne AIDAPS system is installed on the aircraft, the crew chief can use this system as a tool to troubleshoot maintenance problems. The system not only identifies the discrepancy, but it also details the corrective actions by drawing upon the built-in cumulative wisdom of many human experts. This maintenance aid, therefore, decreases the requirements for maintenance diagnostic skills.

It is a well recognized fact that the major portion of troubleshooting time on jet engines is spent in identifying flight discrepancies and trying to duplicate these problems on the ground. If an airborne AIDAPS system is utilized, diagnostic information is immediately available when an aircraft lands which will identify the malfunctioning system component and furnish recommended maintenance actions. In addition, these maintenance actions can be performed by the crew chief and should not require specialist help for pinpointing problems. Maintenance time is thus basically reduced to repair and re-inspect time.

3.3.3 Decreased Spare Parts Use and Operational Cost

In many cases trouble shooting by maintenance people with limited diagnostic skills is done by the process of elimination; i.e., removal of various parts in the system in a sequential manner until the problem is resolved. This type of troubleshooting plus the need to "time change" certain components requires that a large amount of spare parts be kept in readiness. These spare parts stocks and costs will be minimized through the abilities of AIDAPS to accurately diagnose malfunctions and permit implementation of "on-condition" maintenance.

Fuel costs constitute another important part of aircraft operating expenditures. The USAF estimtes that AIDAPS can save 5% on fuel use while commercial airlines generally estimate higher than the 5% figure. This saving is reflected through identifying high fuel flow engines and initiating corrective action. Such savings can be reflected in added flight time or reduced operating costs.

3.3.4 Mission Effect

The airborne AIDAPS system can also provide, if so desired, a cockpit display of real-time information enabling the pilot to judge critical conditions. This will permit the pilot to evaluate the condition versus the mission and take precautionary action when necessary. For example, if a serious condition occurs over a combat zone, the pilot may choose to reduce power or maintain a higher altitude on the trip back to base. If a serious condition occurs shortly after takeoff, the mission may be aborted in the interests of flight safety.

3.4 Prognostic Impact

The ability of the Hamilton Standard AIDAPS system to collect and analyze trending data and perform prognostic functions will produce long range improvements to two difficult maintenance problems that have long hampered aircraft operational availability, both military and civilian. These are TBO (Time Between Overhaul) and Maintenance Scheduling.

3.4.1 Increased TBO (Time Between Overhaul)

Currently, overhaul periods are predetermined based on statistical maintenance data which results in LRU removal from the aircraft on a "time change" basis. Many items are thus removed at prescribed time intervals regardless of the fact that they may still be serviceable and, conversely, degenerating items may not be detected until severe malfunctions occur.

Since the AIDAPS system monitors the performance status of all major LRU's during all phases of flight operations, a constant knowledge of subsystem and component health condition with respect to allowable tolerances is available. This continuous on condition monitoring will reduce time changes to a minimum and will ultimately eliminate time change requirements, thereby, substantially reducing the amount of spare parts required. Thus, an LRU will no longer be removed from an aircraft solely because it has accumulated a prescribed amount of flying time, but rather because it shows signs of wear and impending failure. This on-condition method, when applied to all the aircraft systems, will serve to increase the average time between overhaul (TBO) of the major LRU's in the aircraft and will result in lower spares stocks, lower overhaul costs and high availability.

3.4.2 Improved Maintenance Scheduling

Use of the AIDAPS system will increase (TBO) and give advance notice of overhaul need for all of the major LRU's in the aircraft as explained above. As a result, scheduled maintenance operations will occur less frequently, and can be grouped for an aircraft, thereby permitting maintenance officers to more efficiently plan their scheduled maintenance activities. Assuming that a maintenance officer is working with his present resources, he can more effectively perform scheduled maintenance operations since his specialists are unburdened from the normal trouble shooting functions which the crew chief, with the aid of AIDAPS, can now accomplish. Therefore, due to the more effective utilization of personnel, a reduction in required manpower and training costs is possible to accomplish the same maintenance tasks.

3.5 Program Operation

The entire AIDAPS program was originally planned for completion within a period of one (1) year. The program was divided into four (4) phases as follows:

Phase	Start Date	Completion Date
A	7/1/70	9/30/71
B	10/1/70	11/30/71
C	12/1/70	12/31/71
D	1/1/71	6/30/71

Phases A, B, and C were completed as scheduled, however, Phase D was extended to accommodate a change in scope of the program. Originally, two (2) UH-1 aircraft were to accumulate flying time at Fort Rucker, Alabama, with two (2) AIDAPS systems aboard. Since the flight test location was changed to ARADMAC at Corpus Christi, Texas, it was decided to expand the Phase D program to fly one aircraft with various discrepant components. Also, six nominal sets of components (engine, transmission parts, 42° and 90° gear boxes) were flown to establish baselines for verification testing. This program expansion extended Phase D into the first part of October, 1971.

3.5.1 Location of Work

The following major tasks in the UH-1 AIDAPS program for each phase were accomplished at the following locations:

3.5.1.1 Phase A

1. Fabrication of AIDAPS hardware - HSD
2. Installation Analysis - HSD, Bell Helicopter, and Lycoming

3.5.1.1 Continued

3. Sensor Selection - HSD and Lycoming

4. Engine Sensor Fabrication - Lycoming

5. Generation of Software, PIP and other documentation - HSD

3.5.1.2 Phase B

1. Test Cell Installation - ARADMAC

2. System Calibration - ARADMAC and HSD

3. Test Plans - HSD

4. Test Cell Tests - ARADMAC

5. Test Cell Data Analysis - ARADMAC and HSD

3.5.1.3 Phase C

1. Fabrication of bracketry, etc. - HSD

2. Fabrication of harnessing - ARADMAC and HSD

3. Helicopter Installation and Checkout (2 aircraft) - ARADMAC

3.5.1.4 Phase D

1. Flight Tests - ARADMAC

2. Flight Test Data Analysis - ARADMAC and HSD

3. Calibration of Equipment - ARADMAC and HSD

3.5.2 Manpower Loading at Test Site

The manpower requirements in terms of full time personnel for Phases B, C and D were as follows (no test site activity during Phase A):

Phase B

Four (4) Engineers

1 - Project Engineer (Spent time at test site and HSD as required)

1 - Programmer/Analyst

3.5.2 Continued

1 - Test Engineer (Hardware specialist)

1 - Test Engineer (Vibration specialist)

Two (2) Technicians

1 - Electronics

1 - Transducers

Phase C

1 - Project Engineer

2 - Test Engineers

2 - Technicians

Phase D

1 - Project Engineer (Spent time at test site and HSD as required)

2 - Test Engineers

1 - Technician

It should be noted that throughout the program additional part time manpower was used within budgetary limits to support the program. Additional manpower was needed especially during initial installation of the equipment in the test cell and then on the aircraft.

3.5.3 Role of Bell Helicopter and Lycoming

Bell Helicopter Company (BHC) was chosen as consultant to AVSCOM for the purpose of assisting and advising Hamilton Standard in all technical areas associated with the AIDAPS Test Bed Program. When required, Bell Helicopter Company obtained support from Lycoming. In addition, Lycoming was also selected under separate contract as a vendor to Hamilton Standard to supply

3.5.3 Continued

two sets of engine transducers and harnesses for test cell testing and flight testing. Lycoming also provided to Hamilton Standard a diagnostic software package for adaptation within the airborne software program.

3.5.3.1 Bell Helicopter Company Support

Bell Helicopter Company support during the UH-1 Test Bed Program included the following activities:

3.5.3.1.1 Phase A Installation Liaison Support

During Phase A, Bell Helicopter support was drawn on extensively in establishing various transducer locations throughout the aircraft. Hamilton Standard engineering personnel worked closely with BHC to determine location and method of attachment of transducers. Also, BHC provided all needed drawings, sketches, etc. to aid in the design of various brackets for transducer installations. BHC also assisted in solving problems associated with transducer and AIDAPS hardware installation.

3.5.3.1.2 Phase A Flight Test Plan Consultation

BHC was contacted to aid in the preparation of a Flight Test Plan to be followed during the Flight Test Phase of the program. The combined efforts of Hamilton Standard and BHC were used to generate a Flight Test Plan most suitable for the UH-1 aircraft and its use in the AIDAPS Test Bed Program.

3.5.3.1.3 Diagnostic/Prognostic Data Consultation

BHC efforts were utilized in the refinement of mechanical diagnostic software during the course of the program. BHC reviewed the mechanical diagnostic logic generated by Hamilton Standard and advised changes in certain

3.5.3.1.3 Continued

areas based on their experience with the UH-1 aircraft. Hamilton Standard also suggested changes to the BHC reports.

3.5.3.1.4 Additional Bell Helicopter Company Support

BHC provided the following additional support to Hamilton Standard:

1) Supplied a two-volume data package which provided information required to initiate Phase A activity

2) Reviewed and approved hardware installation plans during Phase A

3) Provided consultation during Phase B data analysis with regard to transmissions and gear boxes.

4) Provided consultation during the Installation Phase C regarding implementation of the approved installation plans for AIDAPS hardware and transducers

5) Provided consultation during Phase D and helped resolve problems encountered during the Flight Test Phase of the program.

3.5.3.2 Lycoming Support

Lycoming support during the UH-1 Test Bed program consisted of assisting Bell Helicopter Company where required to complete support described in Sections 3.5.3.1.1 through 3.5.3.1.4 of this report. In addition, Lycoming supplied two sets of engine transducers and harnesses complete with installation plans and photos for installation in the test cell and UH-1 aircraft. Lycoming also provided a diagnostic software package that was partially used in the final airborne software program. Lycoming also provided support in maintaining this hardware during the Test Bed Program.

3.5.4 Portable Work Area (Van)

3.5.4.1 General Description

The portable work area that was used during Phases B, C, and D of the UH-1 AIDAPS program is a McCarthy Mobile Office No. MC-36, modified to be used as a combination office and laboratory.

The Mobile Office (Van) is 36 feet in length of which 32 feet is usable. The van is divided into two seperate areas (refer to Figure 3-1, 3-1A, 3-1B) designated the Software Office and the Equipment Maintenence and Calibration Office each having their own entrance. The rear portion of the Van houses the equipment required to conduct the software portion of the program including the Digital Processor, Tape Readers, etc. (equipment listed in Section 3.5.4.2). This software section has a separate entrance and office area containing a desk, filing cabinets, and overhead storage cabinets. Likewise, the forward section of the van is suitably equipped to perform hardware service and calibration. The equipment in the van was not built-in but rather the van and the contents were shipped to ARADMAC separately and assembled on site. This approach minimized shipment damage and facilitates disassembly and economical disposition of the van at the completion of the AIDAPS program.

3.5.4.2 List of Equipment in Van

3.5.4.2.1 Software Support Equipment

1 DDP 116 Digital Processor

1 Teletype; ASR33

1 Magnetic Tape Reader

1 High Speed Paper Tape Reader

3.5.4.2.2 Hardware Support Equipment

2 Millivolt Supplies

1 Digital Volt Meter (DVM)

1 Four-Channel Scope and Scopemobile

1 Frequency Counter

1 Function Generator

1 Analog Voltmeter

1 Variable Transformer

1 Megohm Decade Box

Various assortment of Scope Probes, Test Leads, Coaxial Cables, and connectors, etc., to support airborne hardware and calibration activity.

3.5.4.3 Van Power Requirements

The power requirements for the van are as follows:

115 VAC; 60 Hz	-	120 amp service
115 VAC; 3 ∅; 400 Hz	-	750 VA
28 VDC	-	50 amps

The 120 amp service is required to accommodate all the test equipment, lighting, air-conditioning and other utilities. The 115 VAC 3∅ power is required to provide power for the airborne hardware during calibration and maintenance.

The 28 VDC power is required to run the airborne inverters for system testing in the van.

3.5.4.4 Communications

The van comes equipped with standard telephone jacks approved by the Bell System for telephone installation. Two telephones were installed in the van; i.e., one for the Software Office and one for the Equipment Maintenance and Calibration Office. The same extension was used for conference call purposes.

3.5.5 HSD Computer Support

All recorded data that was taken during test cell and flight testing was processed via an on-site ground based computer for preliminary analysis. This computer (the DDP116) was used to confirm that valid information was being obtained for the detailed off-site analysis and verification studies that were completed at the Hamilton Standard computer center.

3.5.5.1 Off-Site Data Analysis

The off-site data analysis utilized an IBM System 370 Computer located at Hamilton Standard. The major tasks that were accomplished by this computer included verification and refinement of the diagnostic logic and data interpretation for trend analysis. The input to the IBM 370 computer was the magnetic tape data obtained from the airborne computer and reviewed by the DDP 116 computer.

The detailed analysis program included the following tasks:

1. Present the recorded data in tabular form in engineering units or the raw data on request.
2. Conduct trend analysis and investigate the feasibility of predicting the life of the components.

3.5.5.1 Continued

3. Store the data and update the trending files as indicated by the trend program.

4. Verify or refine the diagnostic analysis of the airborne computer.

5. Prepare and format the recorded information for final display.

The verification and refinement of the diagnostic logic was an iterative man-machine process. The initial IBM 370 program was a FORTRAN version of the airborne computer program. The airborne-generated action messages were then verified by the ground programs. The tabular data presented by the computer was also analyzed for simplified means of arriving at the same action messages. Second generation programs were devised and checked out to reduce the task of the airborne computer. These second generation programs were incorporated into the airborne computer at opportune times.

3.5.5.2 Data Analysis Flow Chart

The flow chart shown in Figure 3-2 illustrates the procedure followed for each magnetic tape that was generated during the Test Cell Phase B and the Flight Test Phase D.

3.0 Phase A Tasks

Phase A activity was conducted at Hamilton Standard during the period from July 1, 1970 through September 30, 1970. The major tasks involved modification of existing hardware for use on the UH-1 aircraft, a diagnostic logic analysis, installation analysis and sensor selection. Two (2) complete shipsets of hardware were prepared during this phase of the program.

3.0.1 Hardware Modification and Fabrication

The major portion of the hardware used during the UH-1 Test Bed program was redeployed from the Hamilton Standard KSS-DC-9 Airborne Integrated Data Systems (AIDS) program. NOTE: Refer to Section 4.0 for hardware description. The Main Electronics Unit (MEU) and Data Entry Panel (DEP) were modified for adaptation to the UH-1 aircraft. All new signal conditioning circuitry was incorporated in a separate Air Transport Rack (ATR) box called the Auxiliary Box which was interconnected to the MEU and DEP. This auxiliary electronics box was used for expediency recognizing that the MEU had more circuitry than was needed because of the complexity of the DC-9 system. This extra capability could have been removed and the new UH-1 signal conditioning circuitry could have been added to the existing MEU, but this would have resulted in extensive MEU rework. Therefore, it was decided to incorporate all new circuitry into the Auxiliary Box and merely deactivate the circuitry not required in the MEU.

Other hardware was also fabricated specifically for the UH-1. A "brassboard" MAAP (Maintenance Action Annun.Panel) was fabricated (see Section 4.4). Also fabricated were all the brackets, fittings, etc. that were required to mount the sensors and electronics hardware in the UH-1 aircraft. Certain

3.6.1 Continued

items such as the digital magnetic tape recorder did not have to be modified. Also items such as the two (2) 28 VDC to 400 Hz inverters required per shipset were purchased for installation in the UH-1 aircraft. Certain ground base hardware had to be fabricated such as magnetic tape to DDP 116 interface circuitry, etc.

During this phase, Lycoming prepared all the transducers interface hardware that was required for the engine. They also fabricated all engine harnessing that was required. Additional harnessing to interconnect the electronic boxes was also fabricated during this period for test cell and the UH-1 aircraft installations.

All sensors that were selected for use on the UH-1 aircraft were ordered and received during this time period. Certain long lead time items that were not required until the Flight Test Phase were not received until Phase B or Phase C.

3.6.2 Diagnostic Logic Analysis

All government furnished data was reviewed and analyzed and a diagnostic logic analysis was performed. Extensive work was conducted on generating the software that was required for test cell running and the Flight Test Phase. This software was refined during Phase B and further refined during the Flight Test Phase.

During this time period an On-site/Off-site Data Analysis Plan was prepared, and Quick-Look Van analysis equipment was specified. Also a sample simulation tape was prepared. The software that was generated included:

3.6.2 Continued

(a) Recommended parameter limits

(b) Recommended parameter interrelationships

(c) Recommended preliminary diagnostic/prognostic logic

(d) Output data and display formatting

(e) Internal system supervisory program.

3.6.3 Installation Analysis

A detailed installation analysis was performed during Phase A which included trips to Bell Helicopter Company and Lycoming for consultation. All detailed drawings and sketches that were required for equipment installation were prepared during this period. This work culminated in the preparation of the "Flight Hardware Installation Plan for UH-1 Test Bed Program," Document No. H-AIDAPS-0-18, dated 17 September 1970. Details regarding installation are also covered in Section 3.8 of this report.

3.6.4 Sensor Selection

During Phase A, a detailed sensor accuracy analysis was conducted and sensors were selected that would achieve the accuracy requirements of the AIDAPS system. It should be noted that a large number of parameters were chosen using the philosophy that "it is easier to remove items than to later add them." Throughout the program, the contribution of each sensor to the total diagnostic system was reviewed periodically to determine if that particular parameter would be retained or dropped at the end of the program. A detailed sensor and parameter description is given in Section 4.10 of this report.

3.6.5 Other Phase A Activities

3.6.5.1 Program Implementation Plan (PIP)

Besides the activities discussed above, other contract requirements were completed during Phase A such as the generation of a "Program Implementation Plan for UH-1 Test Bed Program", Document No. H-AIDAPS-0-14, dated October, 1970. This plan detailed the Phase B and Phase D test operations.

The Phase B segment included:

(a) Schedule of Activity

(b) Equipment to be tested

(c) Facilities and test equipment to be utilized.

(d) Equipment set-up and sensor installation procedures

(e) Test run procedures

(f) Sensor removal procedures

(g) Documentation plans

The Phase D segment included:

(a) Flight coordination procedures

(b) Flight debriefing procedures

(c) Flight conditions to be evaluated.

3.6.5.2 Procurement of Portable Work Area (Van)

Phase A activity also included the search for a suitable Portable Work Area to be utilized during Phases B, C, and D. It was decided to procure a standard mobile office and prepare the work area on site. This item is discussed in detail in Section 3.5.4 of this report.

3.6.5.3 Initial Airborne System Checkout

Following the fabrication and individual testing of each electronic component, the entire AIDAPS system was checked out during Phase A with certain portions simulated as required prior to shipment of the two (2) systems to ARADMAC. The initial airborne software program was used during the initial system hardware run-up and performance verification.

3.7 Phase B Tasks

Phase B activity consisted of monitoring engines, transmissions, and gear boxes using the two flight hardware systems prepared during Phase A. During this phase, a test goal of 38 engines, 40 transmissions, 40-90° gear boxes, and 24-42° gear boxes was established. The above totals consist of both good (nominal) and discrepant parts. The actual total number of components monitored during Phase B was 28 engines, 63 transmissions, 52-90° gear boxes, and 35-42° gear boxes, for an actual total of 178 components tested against a design goal of 142. A main objective of this phase was to performance test the entire AIDAPS System and prove these techniques could actually detect known discrepant parts.

3.7.1 Test Cell Installation

The test cell installation activity was started on schedule (10-1-70) based on preliminary liaison effort conducted during September between HSD personnel and the ARADMAC Test Cell Engineering Group. Delivery of the McCarthy Mobile Office was accepted in late September 1970. Work to transform the Mobile Office into a suitable portable work area (Van) was initiated during the last week of September 1970 and completed during the first week of October 1970.

3.7.1.1 Test Cell Wiring

The Test Cell wiring installation was designed to minimize the time required for hook-up once a component (engine, transmission or gear box) was installed in its appropriate test cell. Wire bundles were routed through the test cell walls via existing conduit and large electrical quick-disconnect connectors were used at both ends of the wire bundle. This facilitated a

3.7.1.1 continued

quick electrical hook-up of a component (engine, transmission or gear box) that was prepared for test outside the test cell and also enabled quick change of the flight hardware system for calibration, troubleshooting, etc. The harnesses used on each particular component (engine, transmission, or gear box) were also designed for quick change. Each leg of the harness for each sensor location was lengthened and draped over the component with minimum tie-down to the component, thereby expediting the installation and removal of the particular harness.

If a sensor signal was shared between the regular test cell instrumentation and AIDAPS instrumentation, the electrical connection was made at the sensor location. This wiring method minimized the interference with standard test cell instrumentation and expedited troubleshooting of either an AIDAPS System wiring problem or a standard test cell instrumentation wiring problem.

3.7.1.2 Test Cell Sensor Installation

Two sets of sensor hardware were utilized during the test cell phase of the program. This enabled a component (engine, transmission, or gear box) to be prepared for test while another component was being tested in its respective test cell; i.e., sensors were installed and quick-change harness was connected to each sensor outside the test cell. When the test cell was available, the component was mounted in the test cell and the only further requirement was to hook up the electrical quick-disconnect connector to the test cell harness and proceed with the testing. This procedure considerably minimized any interference with the normal activities of the test ce.

3.7.1.3 Electronic Hardware Installation

The electronic AIDAPS equipment including the vibration instrumentation was mounted on a portable cart complete with an interconnecting electrical harness. The harness had a quick-disconnect electrical connector to facilitate quick removal and hook-up to the permanent test cell harness. This feature expedited the removal of AIDAPS electronic hardware and vibration instrumentation from the test cell for checkout in the portable work area (Van). This procedure also made it possible to time share the electronic equipment between the gear box test cells and the transmission test cells, as required.

Upon completion of the test cell testing (Phase B), the test cell harness was left intact for possible future testing during the Flight Test Phase. This proved wise since the test cells were utilized during the Flight Test Phase.

3.7.2 System Calibration

The AIDAPS System has been periodically calibrated to insure that no unknown drift has taken place in the system electronics and sensors. If drift has occurred, it may be trimmed out or accounted for in the data analysis.

After test cell installation and prior to data gathering, the complete AIDAPS system was tested for proper operation, and all hardware and wiring discrepancies were corrected. The vibration instrumentation was set up to be completely independent of the AIDAPS hardware; that is, all vibration sensor inputs were directed to an analog recorder where data was recorded and consequently analyzed as a separate entity. This equipment was also calibrated as a separate entity.

3.7.2.1 AIDAPS Equipment Calibration

All electronic equipment and instrumentation was calibrated twice during the Test Cell Phase. The method of calibration used for the AIDAPS equipment is as follows:

1. Simulate all sensor signals using voltage sources at the inputs to the signal conditioning circuitry in the Main Electronics Unit and Auxiliary Box, which are described in Section 4.2.

2. Select approximately 6 different inputs for each sensor simulation range and input these known signals into the electronics units (MEU and Aux. Box). The output readout is obtained at the Data Entry Panel (see Section 4.3 for description). The output and input values are tabulated for future processing.

3.7.2.2 Sensor Calibration

All sensors used in the system were calibrated once during the test cell phase. These sensors were calibrated as separate entities; i.e., each sensor was sent to its respective calibration lab (pressure, temperature, etc.) and calibrated alone rather than calibrated with the electronic equipment. The method of calibration used for the sensors is as follows:

1. Input actual pressures, temperatures, positions and flows, etc., to each respective sensor via lab standards test equipment.

2. Select approximately 7 different value inputs for each respective sensor and input these physical paramters to the sensor. The output is read in voltage at the output of the transducers. The sensor is electrically loaded to simulate the actual loading conditions in the test cell or aircraft; i.e., the parallel impedance of the electronics unit and aircraft gages would

3.7.2.2 continued

be simulated and the transducer loaded with the simulated impedance during calibration. The output and input values are tabulated for future processing.

3.7.2.3 Calibration Accuracy

A general rule regarding calibration equipment accuracy was adhered to during the calibration of the AIDAPS equipment, sensors, and vibration equipment such that all calibration equipment used should have accuracies greater than airborne equipment accuracies by a factor of 10.

3.7.2.4 Computer Processing of Calibration Data

All calibration data obtained from AIDAPS equipment and sensor calibration was tabulated and transferred to IBM cards. An IBM 370 computer was used to process the calibration data obtained from the AIDAPS airborne hardware. This processing included a program to integrate the calibration data obtained from the AIDAPS airborne electronics and sensors with the data actually obtained from the same during each flight. Therefore, the IBM printout of all parameters that is obtained for each flight is automatically corrected with the latest calibration data that is obtained from either AIDAPS System #1 or System #2.

Rigid control of calibration records was maintained throughout the Test Cell Phase and the Flight Test Phase of the program. For example, every time a sensor was changed or recalibrated, the calibration number was changed on the flight log sheet and a new calibration sheet was forwarded back to HSD. This data was transferred to IBM cards and inputted into the IBM 370 computer when the data for that particular flight was processed. This procedure was also followed when the AIDAPS airborne electronics was recalibrated.

3.7.3 Good and Discrepant Parts Test Plan

A large portion of the data was obtained during normal test cell operation. A review of the normal test run-in procedures indicated that many of the operating conditions and transients are usually investigated and did not require special planning. A certain amount of monitoring of the AIDAPS data was done periodically during the normal component testing while additional monitoring was accomplished as special conditions were obtained during a post test run.

3.7.3.1 Test Procedure for Component Testing

The basic procedure for component testing is as follows:

a. Turn on equipment and perform Van Test (Section 3.7.3.2.1)

b. Prepare the component for sensor installation

c. Fill out HSD data sheet on the component

d. Attach sensors to brackets and install harness

e. Install component in test cell

f. Turn on equipment and perform Test Cell Test (Section 3.7.3.2.2)

g. Run tests per specified Test Plan

h. During steady-state operation, verify that data is entering properly

i. Remove sensors and brackets after component testing

j. Fill in HSD data sheet

k. Verify the data on the magnetic tape with the DDP-116 ground base computer in the Mobile Office

l. Fill in HSD visual displays at Mobile Office at end of day's activities

m. Route magnetic tape back to HSD

n. Revise diagnostic software as a result of data from previous test runs

3.7.3.1 continued

If an abnormal condition occurs during (f) above, an additional period of running time (3 minutes) would be required to allow the development of signature. Note: This extra data was taken only if a safety hazard does not exist.

3.7.3.2 AIDAPS Computer Pre-Test Checkout Procedure

3.7.3.2.1 Check In On-Site Mobile Office

1. Connect MEU, DEP, Auxiliary Box
2. Connect Sensor Simulator to Auxiliary Box
3. Turn on power
4. Interrogate sensors using DEP for correct value
5. Record data on tape
6. Input magnetic tape to Van computer
7. Verify that data was recorded correctly

Note: This check was completed weekly as a minimum.

3.7.3.2.2 Check-In Test Cell

1. Connect all AIDAPS components per installation instructions
2. Turn on power
3. Interrogate selected sensors using DEP; pressures and temperatures will read ambient; speeds, fuel flow, temperature differential will read zero
4. Record data on tape and repeat steps (6) and (7) above

Note: This check was completed before each run. If the equipment appeared to have a malfunction present prior to or after the test cell run, the equipment was returned to the Mobile Office and the test specified in Section 3.7.3.2.1 was performed.

3.7.3.3 Test Cell Test Plan (Gear Box & Transmissions)

The 90° gear box, 42° gear box and transmission tests presented in the U. S. Army Test Data Sheets, WR55-1560-127/-128, WR55-1560-123, and WR55-1560-202/-203 were adequate and were used during test cell testing of the subject components. The only additional testing that was required was a slow RPM sweep of the first five (5) transmissions checked. The purpose of this test was to identify transmission housing resonances.

3.7.3.4 Test Cell Test Plan (Engine)

The engine operating conditions which are tabulated below were monitored during test cell operation. These conditions are needed to confirm the normal and malfunction signatures of the engine for use during the flight test phase. The test conditions have been extracted from Chapter 11 of WR55-2840-113 and the appropriate section is identified with each condition. The sequence of encounter was not critical for AIDAPS. The time at condition as presented in Chapter 11 was adequate. Additional required operation for degraded engine components is specified in Table 3.1.

Engine Condition		Section
1.	Start Up	11-38
2.	Ground Idle	11-39
3.	Military Rated Power	11-54
4.	Flight Autorotation	11-51
5.	Acceleration/Deceleration Transients	11-52
5(a)	Flight Auto. to Military Rated and Return	11-52
5(b)	Ground Idle to Military Rated and Return	11-52

3.7.3.4 continued

Engine Condition		Section
6.	75 Percent Normal Rated Power	11-53
7.	Normal Rated Power	11-54
8.	90 Percent Normal Rated Power	11-44 (Note 1)
9.	Wave Off Test	11-58 (Note 2)
9(a)	Military to 90% N_1 and Return	11-58
9(b)	Military to 75% N_1 and Return	11-58
9(c)	Military to 60% N_1 and Return	11-58
9(d)	Military to 55% N_1 and Return	11-58
10.	Shutdown	11-59

Note 1:

A specific test at 90 percent of Normal Rated Power is not defined in Chapter 11. However, the Vibration Test of Section 11-44 is conducted at several settings of N_1 and N_2 speeds which should cover the desired operation. No additional tests would then be required to cover this condition.

Note 2:

The Wave Off Tests (Section 11-58) are presently conducted at two levels of ambient or P_1 pressure. The AIDAPS monitored both of these pressures initially with the objective of eliminating one of the levels.

3.7.4 Baseline Establishments

The objective during baseline testing was to attempt to establish baselines for nominal LRU's to use as a reference to distinguish discrepant parts from nominal or good parts.

3.7.4 continued

The procedure used in establishing baselines for nominal LRU's was to gather test data on as large a number of engines, gear boxes, and transmissions as was feasible within the scope of the program. The data was utilized as follows:

1. To determine whether normalized baseline data or customized baseline data could be used during the flight test program; i.e., normalized data being an average band of values for each parameter obtained from testing a large sample of components while customized data is the data taken for each parameter from testing each specific component.

2. To determine the nominal spread of data for each component; i.e., engine, transmission, 42° gear box, and 90° gear box.

3. To determine by comparison of the baseline data taken in the Test Cell and baseline data taken in the Flight Test Phase what baseline data to use in a deployable system.

3.3 Phase C Tasks

Phase C activity consisted primarily of flight hardware installation on two (2) UH-1 aircraft. This activity extended for a period of approximately one (1) month. All hardware that could be prefabricated was fabricated at HSD (brackets, fittings, etc.) prior to the beginning of Phase C. Since no mockup of the UH-1 was available, a large portion of fabrication was completed at ARADMAC (harnessing, etc.). All modifications to the UH-1 aircraft were made without affecting major sub-assemblies.

This section of the report outlines details of sensor and equipment installation. The actual installation drawings and bracket detail drawings are not included due to the quantity of drawings involved. However, reference to drawing numbers for each sensor and equipment installation is given in Tables 3.2, 3.3, 3.4, 3.5, and 3.6 of this report. Every drawing listed is available upon specific request. Each specific sensor installation is identified by an HS item number. The tables listed above tabulate all drawings required for each specific HS item number.

A system wiring diagram was also prepared (Ref. SK 79365, Wiring Diagram, System Cabling, UH-1, AIDAPS). This diagram details all wire routing throughout the aircraft.

3.3.1 Sensor Installation

The sensors that are mounted throughout the aircraft are separated into four separate groups. Each group of sensors has separate harnesses which are routed along existing aircraft wiring bundles where possible. The four groups are the Tail Boom Sensor group, the Engine Sensor group, the Transmission and Hydraulics Sensor group, and the Instrument Panel Sensor group.

3.8.1.1 Tail Boom Sensor Group

The Tail Boom Group is composed of the following various sensors located in the 42° gear box, the 90° gear box, and along the tail rotor drive shaft. Installation drawings and associated detail drawings for each sensor are listed in Table 3.2 of this report.

HS Item No.	Parameter	Section
54	#2 Hanger Bearing Vibration	Tail Rotor Drive Shaft
56	#3 Hanger Bearing Vibration	Tail Rotor Drive Shaft
104	#4 Hanger Bearing Vibration	Tail Rotor Drive Shaft
59	Input Quill Vibration	42° Gear Box
61	Output Quill Vibration	42° Gear Box
58	Δ T Oil Temperature	42° Gear Box
90	Chip Detector	42° Gear Box
64	Input Quill Vibration	90° Gear Box
66	Output Quill Vibration	90° Gear Box
63	Δ T Oil Temperature	90° Gear Box
91	Chip Detector	90° Gear Box

The Tail Boom Sensor wiring is routed along existing wire as per detail shown in Drawing SK 79730-160. The direction of the routing is as follows:

The 90° gear box wire bundle merges with the 42° gear box wire bundle and the combined bundle runs along the right hand side of the tail boom picking up wiring from #2, #3, and #4 hanger bearings along the way. After passing the tail boom attach bulkhead, this wire bundle runs along the right

3.8.1.1 continued

side of the aircraft and merges with the engine harness underneath the deck behind the right side of the rear engine firewall bulkhead.

3.8.1.2 Engine Sensor Group

The Engine Sensor Group is composed of the following sensors located on the engine and in the engine compartment. Installation photographs and associated detail drawings for each sensor are listed in Table 3.3 of this report.

HS Item No.	Parameter	Section
1	#2 Bearing Oil Scavenge Pressure	Engine
2	#2 Bearing Oil T Temperature	Engine
3	#2 Bearing Chip Detector	Engine
5	Average EGT	Engine
6 (a-l)	EGT Pattern	Engine
9	#3/4 Bearing Oil Scavenge Pressure	Engine
10	#3/4 Bearing Oil Δ T Temperature	Engine
11	#3/4 Bearing Chip Detector	Engine
12	Acc. Gear Box Chip Detector	Engine
13	P_3	Engine
14	T_3	Engine
15	P_1	Engine
16	T_1	Engine
17	N_1	Engine
18	N_2	Engine
19	PLA	Engine

3.8.1.2 continued

HS Item No.	Parameter	Section
20	W_f	Engine Comp.
21	Left Fuel Boost Pump Flow Switch	Engine
22	Right Fuel Boost Pump Flow Switch	Engine
23	Fuel Pressure	Engine
24	Eng. Driven Fuel Pump Pressure Switch	Engine
26	Fuel Filter Δ P Switch	Engine
29	Lub. Oil Temperature	Engine
30	Lub. Oil Pressure	Engine
31	Oil Filter Δ P	Engine
32	Torque	Engine
34	Bleed Band Position	Engine
36	Starting Battery Volts	Engine
37	Engine Ignition Exciter	Engine
119	IGV Angle	Engine
120	Inlet Air Filter Δ P Switch	Engine
121	Engine Oil Pressure Switch	Engine
122	Fuel Temperature	Engine
52	#1 Hanger Bearing	Engine Comp.
4	Combustion Flange Vibration	Engine
7	#3/4 Bearing Scavenge Line Vibration	Engine
8	Inlet Housing Vibration	Engine

3.8.1.2 continued

The engine wiring harness was pre-fabricated and all wires terminated at three firewall connectors which were mounted on the lower aft firewall. The wire bundles are routed from the firewall to a compartment underneath the righthand engine maintenance deck via a 6" inspection hole located in the deck just aft of the lower aft firewall. Beneath the maintenance deck, the three engine harness wire bundles merge with the tail boom harness previously described, and then the combined harness continues along the right side of the aircraft following the heater duct under the right engine maintenance deck. Entrance to the cabin area is made via the heater duct near A/C Station #156 at the aircraft floor line. This heater duct is no longer used on the UH-1H aircraft.

3.8.1.3 Transmission and Hydraulics Sensor Group

The transmission and hydraulics sensor group is composed of the following sensors located in the transmission compartment and behind the Station #129 bulkhead. Installation drawings and associated detail drawings for each sensor are listed in Table 3.4 of this report.

HS Item No.	Parameter	Section
38	Oil Pressure Switch	Transmission
39	Oil Pressure	Transmission
40	Oil Temperature Switch	Transmission
41	Oil Temperature	Transmission
43	Oil Sump Chip Detector	Transmission
51	External Oil Filter ΔP	Transmission
101	Oil Cooler Flow	Transmission

3.8.1.3 continued

HS Item No.	Parameter	Section
102	Main Rotor RPM	Transmission
103	Internal Oil Filter Δ P	Transmission
45	Input Quill Vibration	Transmission
47	Upper Mast Vibration (axial)	Transmission
49	Tail Rotor Quill Vibration & Sump Input Quill Roller Bearing Vibration	Transmission
123	Upper Mast Vibration (radial) Vibration	Transmission
125	Input Quill Gearmesh Vibration	Transmission
128	Offset Accessory Drive Gearmesh Vibration	Transmission
126	Tail Rotor Quill Gearmesh Vibration	Transmission
127	Upper and Lower Sun Gearmesh Vibration	Transmission
129	Hydraulic Pump & Tach Quill Ball Bearing	Transmission
69	Pressure Switch	Hydraulics
70	Supply Pressure	Hydraulics
71	Supply Temperature	Hydraulics
72	Filter Δ P	Hydraulics
107	Pump Case Leakage Flow	Hydraulics
108	Pump Case Leakage Flow Δ T	Hydraulics

The transmission harness is routed along existing wire bundles and clamped at various points in the transmission compartment as required. The hydraulics harness is routed behind the panels at the Station #129 bulkhead and merges with the transmission harness at the floor line of the Station #129 bulkhead. This harness enters the cabin area via an existing bulkhead connector opening

3.8.1.3 continued

at the floor line of the Station #129 bulkhead. There are existing cutouts at this location where bulkhead connectors for the armament system are mounted.

3.8.1.4 Instrument Panel Sensor Group

The Instrument Panel Sensor Group is composed of the following sensors located behind the instrument panel and below the lefthand control stick. Installation drawings and associated detail drawings for each sensor are listed in Table 3.5 of this report.

HS Item No.	Parameter	Section
77	Total Pressure	Inst. Panel
78	Static Pressure	Inst. Panel
87	Overspeed Governor Switch	Inst. Panel
109	28 VDC Essential Buss	Inst. Panel
110	115 VAC Essential Buss	Inst. Panel
111	26 VAC Inst. Buss	Inst. Panel
112	Hi/Lo RPM Warning Light	Inst. Panel
113	Hi/Lo RPM Warning Audio	Inst. Panel
80	Collective Pitch Stick Position	L.H. Control Stick

The instrument panel harness is routed along existing wire bundles behind the instrument panel and runs up the wire bundle between the left and right windshields and along the roof between the overhead consoles. Wiring from the collective pitch synchro mounted under the lefthand control stick is also included in this wire bundle. The wire bundle enters the cabin area from overhead at the Station #129 bulkhead and is routed through an existing roof opening down the bulkhead to the seat equipment platform.

3.8.2 Fabrication of Aircraft Harnessing

Since pre-fabrication of aircraft wiring harnesses is not feasible without the aid of a mockup, all UH-1 sensor wiring harnessing, with the exception of the engine harnesses, was fabricated on site. The method of fabrication was to start at the most remote individual sensor connector in a sensor group and the proceed to systematically build up the harness by merging with each sensor in the sensor group. Eventually all sensor wiring terminated in various wire bundles at three locations in the rear of the aircraft cabin; i.e., the righthand heater duct at Station #166, at bulkhead connectors near the floor line in the center of the aircraft at Station #129 bulkhead, and from the overhead at the Station #129 bulkhead. All harnessing was tied down at various points via cable clamps and nylon cable straps as required for a secure installation. Wherever possible, wires were routed along existing wire bundles as shown in Drawing SK 79730-160 (Aircraft Wire Routing Method).

3.8.3 Electronic and Instrumentation Equipment Installation

All electronic hardware "black boxes," vibration instrumentation equipment, and inverters are mounted to plywood platforms which are securely mounted to the passenger seats at the rear of the cabin composed of the five-man seats across the aircraft at Station #129 and both two-man seats on the left and right side of the transmission compartment. The equipment layout is detailed in the equipment installation drawing SK 79850-1. The associated detail drawings for this equipment are listed in Table 3.6 of this report. All wiring from the aircraft is routed along the transmission

3.8.3 continued

bulkheads at the floor line under the seats to the connectors at the rear of the electronic and instrumentation equipment. Harnessing that interconnects the electronic equipment and vibration instrumentation equipment was pre-fabricated at Hamilton Standard.

3.8.3.1 Electronic and Instrumentation Equipment Power Source

The power source for all electronic and instrumentation equipment is two inverters mounted as shown in the equipment installation drawing SK 79850-1. D.C. power required to drive the inverters is obtained directly from the non-essential 28 VDC buss in the left aft electrical compartment. The two size #8 wires from the inverters were routed through the heater duct near Station #166 on the lefthand side of the aircraft at the floor line. From this point the wires were clamped to the existing wire bundle that runs to the left aft electrical compartment. Termination was via wire terminals at the non-essential 28 VDC buss. Fifty (50) amp capacity circuit breakers were used in series with the inverter leads.

3.8.4 System Checkout

Upon completion of the sensor installation, equipment installation, and harnessing, the complete system was continuity inspected and a comprehensive "power on" check was performed. Sensor inputs were simulated to complete a system dynamic checkout.

Prior to ground run-up, a representative from AVSCOM inspected the aircraft for compliance with flight safety requirements and adherence to the installation plans submitted during Phase A of the program. Following ground run-up, the aircraft was flight tested to ensure proper performance of the entire system prior to data gathering.

3.8.5 Proof of Non-Interference with Existing Aircraft Systems

The method used to eliminate interference problems with existing aircraft systems was to parallel existing aircraft signals at their source rather than splicing in at aircraft junction boxes. This method required extensive cabling, but expedited troubleshooting of the system when it became necessary to determine whether AIDAPS equipment or existing aircraft hardware and/or wiring was causing a particular problem because a "Y" electrical connection was utilized at sensor connectors which provided a means to disconnect AIDAPS wiring to isolate a particular problem.

All existing sensor outputs used by the AIDAPS system were terminated into impedances above the minimum impedance levels specified by Bell Helicopter Company in order to maintain accuracy of the UH-1 instrumentation. This eliminated the possibility of electrically loading the existing aircraft signals that were also utilized by the AIDAPS system. During the checkout procedure, all existing sensor signal outputs that were paralleled with AIDAPS signal inputs were checked for non-interference. This was accomplished by disconnecting the AIDAPS signal lead and insuring that no change in the existing aircraft signal was present.

3.0 Phase D Tasks

Phase D activity consisted of a 9-1/2 month long flight test program utilizing two airborne AIDAPS systems installed in two UH-1 aircraft. One aircraft #17223 was deployed to gather trend data based on using all original "zero time after overhaul" parts. This aircraft accumulated a total of 245 flight hours. The other aircraft, #61011, was utilized as a discrepant parts test aircraft where discrepant parts were implanted in major LRU's (engines, transmissions, and gearboxes) and the aircraft then flown with these discrepant LRU's. Baseline data was also obtained with #61011 utilizing nominal LRU's. Originally, the flight test portion of the program (Phase D) was scheduled to be completed within a six month period. However, the scope of the program was enlarged to include flying discrepant parts and gathering additional baseline data on six additional sets of LRU's. This addition to the program extended Phase D from a completion date of June 30, 1971 to October 12, 1971.

Verification Testing was also completed during this phase of the Test Bed Program during which unknown discrepant parts were implanted in various LRU's installed in either aircraft and then test flown. The results of these verification tests are discussed in Section 7.0 of this report. The airborne software was constantly being refined, as required, as the Flight Test Program progressed.

At the completion of Verification Tests, AIDAPS hardware was entirely removed from aircraft #17223 and the aircraft restored to its original condition. The remaining baseline testing was accomplished using six sets of LRU's on aircraft #61011.

3.9.1 Phase D Flight Test Procedures

Certain procedures and specific flight profiles were followed during the Flight Test Program involving the nominal and discrepant LRU's that were flown in aircraft #61011. These procedures were also followed for data that was obtained on the trend aircraft #17223. The following paragraphs detail the procedures and flight profiles used during Phase D of the Test Bed Program.

3.9.1.1 Flight Procedures and Debriefing

3.9.1.1.1 Preflight

The following outline was incorporated as part of the standard preflight procedure:

1. Verify that sufficient magnetic tape is available in the analog and digital recorders.
2. Turn on system electrical power.
3. Confirm status of diagnostic program.
4. Enter aircraft and documentary identification information.
5. Clear MAAP.
6. Log counter numbers.
7. Put AIDAPS in "Standby."
8. After engine start-up, monitor parameters specified on Flight Log Sheets using Data Entry Panel.

Note: Periodically perform complete AIDAPS check using sensor simulator, manual insertion panel, etc.

3.9.1.1.2 During Flight (A/C #17223 only)

1. Encounter conditions specified in Flight Profile (see Section 5.2.6).
2. Maintain condition then depress "EVENT" button.

3.9.1.1.2 continued

3. Depress "EVENT" button just prior to a transistion.

4. Depress "EVENT" button whenever a special event occurs which may be of interest,

5. Record reason for "EVENT" indication on Flight Log Sheet and whether event did actually occur.

3.9.1.1.3 Post-Flight

1. Record status of MAAP indicators.

2. Record status of mechanical counters.

3. Remove magnetic tapes and mark reel for identification.

4. Turn off power.

5. Obtain copy of Flight Log Sheets.

6. Review flight with crew for indication of special events.

3.9.1.2 Flight Profile

The flight profile used during Phase D of the Test Bed Program is explained in detail in Section 5.2.6 of this report.

3.9.2 Baseline and Known Discrepant Parts Testing

The Baseline and Known Discrepant Parts Testing accomplished during Phase D was conducted according to the sequence outline in Table 3.7. The policy followed during discrepant parts testing was to fly discrepant engines and 42° or 90° gearboxes together with a good transmission. Discrepant transmissions were flown with good engines and gearboxes. This policy minimized the danger of multiple failures during flight.

Four sets of LRU's were simultaneously utilized in groups of two to obtain the discrepant part test data listed in Table 3.7. Discrepant parts

3.9.2 continued

listed in Table 3.7 were implanted in two sets of LRU's while the other group of two sets was being test flown with discrepant parts already installed. The two (2) just built-up sets of LRU's were run through the test cells and then installed on the aircraft assigned for discrepant part testing for each vendor. While these aircraft were being flown by each vendor, the two other sets of LRU's removed from the aircraft were being built-up in various assembly shops. In this way four sets of LRU's each having different discrepant part combinations were continually cycled between the two vendor aircraft and the assembly shops. These four sets of LRU's were finally re-assembled into their original configuration and flown to obtain baseline data.

3.9.3 Unknown Discrepant Parts Testing (Verification Testing)

Verification Testing was conducted following the completion of the baseline and known discrepant parts testing discussed in the previous section. Unknown discrepant parts were implanted series of six sets of LRU's (engines, transmission parts, 42° gearbox and 90° gearbox) in a sequence that provided for both vendors to alternately obtain data and allowed for build-up time in the various assembly shops. Following the test flights with all six sets of LRU's, the LRU's were re-assembled to their original configuration and all six sets flown to obtain baseline data. The completion of baseline testing on these six sets of LRU's marked the conclusion of Phase D of the Test Bed Program.

A complete discussion of the discrepant parts found during Verification Testing is covered in detail in Section 7.0 of this report.

3.9.4 Baseline Establishments

Baseline data was thus gathered on a total of ten sets of LRU's during the Flight Test Phase of the Test Bed Program to determine the nominal spread of data for each subject LRU: engine, transmission, 42° gearbox, and 90° gearbox. This data was compared to the baseline data that was gathered during the test cell runs (Phase B) to determine what baseline data to use in a deployable system.

It was determined that the baseline data gathered during flight testing did differ from test cell baseline data, especially in the area of vibration recording. The discussion of the differences are discussed in detail in Sections 5.0 and 6.0 of this report.

It should be noted that the main transmission assembly was not changed during baseline data flights. The original transmission was used and different nominal sub-assemblies selected from among the main mast bearing, input quill, and output tail rotor quill were installed for each one of the ten baseline flights.

SECTION 4.0

AIRBORNE HARDWARE

4.0 AIRBORNE HARDWARE

4.1 Airborne Electronics Block Diagram

A block diagram of the Airborne Electronics is shown in Figure 4-1, and a corresponding photo of the electronic packages is shown in Figure 4-2. Figure 4-1 indicates communication paths among each of the electronics packages. The descriptions and functions of the packages are detailed in the following paragraphs.

4.2 Main and Auxiliary Electronic Unit

A block diagram of the electronics is shown in Figure 4-3. In keeping with the objective of "off-the-shelf" hardware, a second or auxiliary package (Figure 4-4 and 4-5) was used to house additional electronics involved in the revision of an existing Main Electronics Unit (MEU)(Figure 4-6). Therefore, discussion of the electronics inherently involves both the Main and Auxiliary packages. The physical size of each package is the same: 1 ATR long, ARINC 404, including Supplement #2.

Input signals are broken into three basic types; Discretes (i.e., the on-off position of a switch); Analog (i.e., a voltage level either AC or DC); and Vibration (i.e., a combination of many frequencies of alternating voltages). The vibration portion will be handled separately in Paragraph 4.7. Discretes, being digital in nature (i.e., a bi-level DC voltage) need only be interfaced to proper voltage levels before being inputted into the computer. The analog signals, having similarity, are broken down into five input groups based on similarity.

1. AC Voltage Ratio
2. DC Differential Voltage

4.2 continued

3. Synchro Voltage

4. Thermocouple Voltage

5. Flows or Speed Frequencies

As shown in Figure 4-3, multiplexing techniques are used which enable all the signals of one type to be switched through the same signal conditioner at a different point in time (time-sharing). This addressing by the digital processor to the electronic switches is broken into a basic timing pulse of 10 milliseconds for a capability of 200 switch closures in a two-second frame, after which the process starts over again.

The function of the signal conditioners is to receive a signal corresponding to the five different types of inputs and convert it into a standard voltage level over the range of -5 to +5 volts DC. Hence, all analog signals appear as a DC level at the output of the signal conditioners. These so-called "time-shared" signals are then switched through the analog to digital converters which convert a DC voltage to a group of coded pulses that is inputted into the digital processor. The digital processor also controls the output interfaces which go to the Data Entry Panel, the Maintenance Action Annunciator Panel, and the Digital Recorder. The following paragraphs details each one of these units.

4.3 Data Entry Panel (DEP)

A photograph of the Data Entry Panel is shown in Figure 4-7. It is approximately 4" x 5" x 7" (L x W x H) and was an "off-the-shelf" unit. It was designed in conjunction with the Main Electronics Unit to provide manual

4.3 continued

control over the digital processor and digital recorder. Controls available are a function of the [illegible] the digital processor, but basically involve recording data at different rates and writing pertinent information on tape using the ten numerical pushbuttons. The pushbuttons can also be used to call up on the eight digit display, any parameter or sensor signal the operator may want to observe. This capability was used for on-the-spot engineering information without having to wait for the data to be processed from the tape. The DEP was thus a valuable engineering tool during the Flight Test Program.

4.4 Maintenance Action Annunciator Panel (MAAP)

A photograph of the MAAP is shown in Figure 4-8. It consists of 64 BITE (Built-In-Test-Equipment) indicators that are electromechanical devices having a black ball which rotates into a locked position and indicates white when triggered by digital processor in the Main Electronics Unit. In this way, a permanent record was made (until manually reset) of a malfunction or of an aircraft parameter when it exceeded prescribed limits as determined by the digital processor from the sensor signals inputs. A crewman can then refer the BITE number to a diagnostic message and perform indicated inspection procedure or maintenance action as designated by the message.

Counters are also contained in the MAAP which were used to register the number of data frames recorded by the Digital Recorder and number of seconds in flight. This MAAP was designed as a "brassboard" unit specifically for the Test Bed Program, and a final configuration would be considerably smaller in physical size.

4.5 Digital Recorder

Photographs of the Digital Recorder are shown in Figures 4-9 and 4-10. It is an "off-the-shelf" item that physically is 1 ATR long, ARINC 404, including Supplement #2. Contained in the unit is a seven track digital [illegible] rea[illegible] a power [illegible] operates from the 115 VAC 400 Hz source. The [illegible] contained in a readily [illegible] on top of the recorder. This tape system is compatible with the IBM system 370 that was used to analyze the data at Hamilton Standard.

4.6 Maintainability

The AIDAPS system, utilizing its on-board processor, has the built-in capability of checking its own internal operation for such things as limit errors in drifts of amplifiers and signal conditioners, as well as giving immediate indication of an abnormal exceedance in limits of any sensor signal. This communication is accomplished with indicators on the Data Entry Panel; and, by pushing the proper button, the display can show what has exceeded specification either internally or externally to the electronic system.

4.7 Vibration Equipment

As shown in Figure 4-1, the vibration recording hardware was not integral with the basic AIDAPS. For the purposes of the Test Bed Program only, conventional analog vibration monitoring and recording equipment was assembled and mounted for temporary airborne use. This instrumentation is fully described in Section 6.4.1.

4.3 Hardware and Sensor Installation Photos

Several photos of the AIDAPS hardware and sensor installations were taken during the test cell and flight test phases. These photos follow to illustrate a sample of the hardware used during the program and the method of installation.

Figure 4-11 - A 42° gearbox and a 90° gearbox being monitored in the test cell at ARADMAC.

Figure 4-12 - A cre [illegible] itored in the test cells at ARADMAC.

Figure 4-13 - Temporary vibration recording equipment used during the test in all phases.

Figure 4-14 - Engine being monitored in the test cells at ARADMAC.

Figure 4-15 - Quick change engine harnessing used in the test cell at ARADMAC.

Figure 4-16 - Manifold in series with oil scavenge lines. Used to monitor #2/#3, 4 bearing oil Δ T, Δ P and chip detection.

Figure 4-17 - Transmission oil cooler flow meter installation on the UH-1H aircraft.

Figure 4-18 - Fuel flow and temperature sensor installation on the UH-1H aircraft.

Figure 4-19 - Aircraft installation of the IGV synchro for the L13 engine.

Figure 4-20 - Hydraulic filter Δ P installation on the UH-1H aircraft.

Figure 4-21 - Bleed band switch on L13 engine.

4.8 continued

Figure 4-22 - Collective pitch synchro installation on UH-1H aircraft.

Figure 4-23 - Hydraulic bypass ΔT and flowmeter installation on UH-1H aircraft.

Figure 4-24 - Aircraft installation of the delta temperature for the 90° gearbox.

4.9 Aircraft Parameters Monitored

This section will discuss the actual parameters monitored. The sensors respectively used to monitor the subject parameters are discussed in Section 4.10 of this report.

4.9.1 Power Plant Parameters

The engine parameters monitored are discussed in this section. Refer to Figures 4-25 and 4-26 for engine station diagram and location of the parameters monitored.

4.9.1.1 Compressor Inlet Total Temperature and Total Pressure (T_{t1} and P_{t1})

These are two of the most important parameters that were measured. The range of values of these parameters form the boundary condition for the evaluation of the engine thermo-dynamic model. They provide the basis for evaluation of engine performance and are commonly known as the "face" of the engines. All of the thermodynamic performance parameters in the gas flow path are related to and influenced by the values of the compressor inlet parameters.

4.9.1.2 Compressor Discharge Total Temperature (T_{t3})

The compressor discharge total temperature is one of the variables used in calculations of the thermodynamic model. It is a factor in the evaluation of compressor efficiency and the gas producer turbine inlet temperature.

4.9.1.3 Compressor Discharge Pressure (P_3)

The compressor discharge pressure is also measured because it is one of the variables used in calculations of the thermodynamic model. It is a factor in the evaluation of compressor efficiency and the gas producer turbine inlet temperature.

4.9.1.4 Gas Producer Speed (N_1)

The rotational speed of the gas producer is fundamental to engine performance calculations. The referred value of the gas producer speed is used in several calculations such as high-low limit, malfunction detection and diagnosis, and trend analysis.

4.9.1.5 Power Turbine Speed (N_2)

Power turbine speed is fundamental to engine power output. When considered along with power lever angle, fuel flow rate, and torque-meter pressure, the performance with respect to shaft power output can be evaluated. The power output or change in power output, when considered along with changes in other parameters, forms the basis for malfunction detection and diagnosis.

4.9.1.6 Fuel Flow Rate (W_f)

Fuel flow rate is fundamental to the evaluation of the engine thermodynamic model. For every power settting and set of environmental conditions, there exists a proper fuel consumption rate. Almost all efficiency changes in the engine components cause resulting changes in fuel consumption. Therefore, changes in fuel flow, when compared to or plotted against other parameters, indicate the area of degradation.

4.9.1.7 Fuel Temperature

Fuel temperature is measured along with fuel flow rate in order to determine the density of the fuel. The density of fuel is used to calculate mass rate of flow which is the parameter that is most useful.

4.9.1.8 Engine Driven Fuel Pump Pressure Switch

A pressure switch is used to measure the engine driven fuel pump pressure in the fuel control. When the output pressure drops below a predetermined value, it is an indication of a discrepant fuel pump, which would result in a fuel control change.

4.9.1.9 Fuel Filter Δ P Switch

A Δ P switch is placed across the fuel filter in the fuel control for the purpose of determining if the fuel filter is clogged. This switch is used to display a diagnostic message "clean fuel filter" before the differential pressure reaches a dangerous level.

4.9.1.10 Exhaust Gas Total Temperature (EGT) (T_{t9})

Exhaust gas total temperature gives an indication of the energy level of the gases through the turbines. It must be examined for a high limit value to prevent over temperature operation. It is used in the thermodynamic model as a dependent variable to establish component performance coefficients. Both individual thermocouple readings and an average reading of all thermocouples in the harness were monitored in order to locate hot spots in the engine.

4.9.1.11 Power Lever Angle (PLA)

The power lever angle is important as a means of correlating a standard performance level with that actually obtained from the engine. The power lever angle is the indicator for basic power setting. When related to engine pressure ratio, fuel flow and engine rotor speeds, the existence (or non-existence) of a proper correlation is indicated. The high-low checks or any other check are valid only if predetermined power settings are established.

4.9.1.12 Inlet Guide Vanes (IGV)

The inlet guide vanes position was also used to correlate a standard performance level with that actually obtained from the engine. The measurement of inlet guide vane position was used to clarify the results of the gas path analysis performed on the engine.

4.9.1.13 Compressor Interstage Bleed Band Position

The compressor bleed band bleeds air from the final stage of the axial flow compressor section and is automatically controlled. It is normally open during engine operation below 70% of N_1 speed or during engine acceleration. The position of this valve (actuator) was monitored to determine proper bleed band operation.

4.9.1.14 Torquemeter Differential Pressure

The torquemeter operates to provide a differential oil pressure which is proportional to the torque applied to the power turbine reduction gearing. A torque value is obtainable from this differential pressure. From the torque pressure and the speed of the power turbine, the horsepower developed can be determined. This is a most desirable factor for the engine thermodynamic model evaluation.

4.9.1.15 Engine Ignition Exciter Voltage

The engine ignition exciter voltage was monitored as a discrete signal to fault isolate problems in the engine ignition system.

4.9.1.16 Engine Oil Temperature

Oil temperature provides information which relates to incipient bearing failure including knowledge of heat added by malfunctioning components. Oil

4.9.1.16 Continued

temperature will provide information regarding mechanical friction increases which is pertinent to trend analysis.

4.9.1.17 #2/#3, #4 Bearing Oil Δ T

As indicated above, oil temperature can provide valid data of incipient malfunction. To obtain a valid diagnostic tool from oil temperature, use is made of the oil temperature out as related to oil temperature in would be the two parameters which measure the increase in frictional loads. This measurement was made by measuring oil temperature into the bearings and measuring oil temperature at #2 and #3/4 scavenge lines.

4.9.1.18 Oil Pressure

Proper oil pressure is essential to proper engine lubrication and cooling. The oil pressure should be examined for a high or low value to detect clogging or leaking. When recorded over a period of time, it can give indications of lubrication system component degradation. An engine oil pressure transducer was utilized along with an engine oil pressure switch to respectively measure system oil pressure for trending purposes and to provide a discrete signal for warning of a "below minimum" value of oil pressure. Pressure switches are also incorporated in the #2 and #3/4 bearing oil scavenge lines to determine if leakage is present at the bearing seals.

4.9.1.19 Chip Detectors

The T53 engine as installed in the UH-1H is equipped with a chip detector located in the accessory case sump. While such a detector can indicate the

4.9.1.19 Continued

accumulation of chips, it does not locate the source. To segregate the sources, the installation of two additional chip detectors in the #2 and #3/4 bearing oil scavenge lines was accomplished. As shown in Figure 4-26 Chip Detector #1 (CD#1) collects chips from the forward components. (CD#2) was located in the external oil scavenge line leading from the number two bearing. (CD#3) was located in the external oil scavenge line leading from the power turbine bearings. A signal from any one of these chip detectors would indicate component degradation, be considered a "no-go" condition, and locate the area of difficulty.

4.9.1.20 Oil Filter Differential Pressure

The pressure drop across the oil filter was examined for a high-low limit. A high value indicates a dirty filter for which the system will initiate maintenance instructions.

4.9.2 Transmission Parameters

The transmission parameters monitored are discussed in this section. Refer to Figure 4-27 for location of parameters monitored.

4.9.2.1 Oil Temperature

An increase in oil temperatures will discern mechanical friction increases occurring in the transmission and thus provide valuable data as to incipient malfunction and prediction of on-condition removal as well as identify degradation of the oil pump and cooling system. An oil temperature transducer and an oil temperature switch has been provided for this purpose.

4.9.2.2 Oil Pressure

Oil pressure should remain constant for the life of the transmission. Decrease in pressure could mean seal leakage or clogged filter; low pressure and high temperature would mean insufficient oil; increase in pressure and temperature could mean malfunctioning of the oil pump or the cooling system. These signals are oriented towards preventive maintenance action at the first maintenance echelon. An oil pressure transducer and an oil pressure switch was provided for this purpose.

4.9.2.3 Filter Pressure Differential

The pressure drop across the oil filter was examined for a high limit as in the case of the engine oil filter. A high value indicates a dirty filter for which the system will initiate maintenance instructions.

The two filters that are monitored by ΔP transducers on the UH-1 transmission are the internal oil filter and the external oil filter.

4.9.2.4 Chip Detector

Suspension of significant foreign matter in the transmission lubrication system can cause catastrophic malfunction. When it is detected, an examination of the oil and identification of the source of the "chip" is mandatory so that the responsible unit, if any, may be removed and replaced.

4.9.2.5 Oil Cooler Flow

Knowing the condition of the transmission at all times is important to safe helicopter operation. The condition of the lubrication system is a basic indicator of transmission health, so that important indicator and knowing the reason for an increase in oil temperature can determine if a major or minor malfunction may be developing in the transmission.

4.9.2.5 continued

A flowmeter has been installed between the thermal control valve and the oil cooler to determine the amount of oil flow through the cooler at any time. This parameter when compared with oil temperature can isolate problems in the oil cooling system; i.e., high oil temperature and high oil cooler flow can mean an inefficient oil cooling system or excessive friction in the transmission. If the accumulated time between oil changes is large, the problem is probably an inefficient oil cooling system, clogged oil cooler, etc. If the accumulated time is low, an excessive friction problem may be present. A high oil temperature and a low oil flow can mean a defective thermal control valve. A low oil temperature and a high oil cooler flow can also mean a defective thermal control valve.

4.9.2.6 Main Rotor RPM

Main rotor RPM is monitored to determine if the rotor is in an underspeed or an overspeed condition. Rotor speed is also compared to the audio and visual rotor speed warning system signals to differentiate between rotor speed warning system problems and actual rotor speed problems.

4.9.3 Gearbox Parameters

Both 42° and 90° gearbox parameters are discussed in this section and their locations are shown in Figures 4-28 and 4-29, respectively. Parameters for 42° gearbox and 90° gearbox are identical.

4.9.3.1 Oil Temperature ΔT

The difference between the outside air temperature in the vicinity of the gearbox and the actual gearbox oil temperature is monitored using two thermocouples. The temperature rise in the gearbox oil is important in determining if excessive friction exists either in the 42° or 90° gearbox.

4.9.3.2 Chip Detector

A chip detector is used in both the 42° and 90° gearboxes to detect if metal chips are present in the gearbox, thereby providing a warning of an incipient gearbox failure.

4.9.4 Hydraulic System Parameters

4.9.4.1 Hydraulic Supply Pressure

Hydraulic system pressure is monitored to determine the condition of the hydraulic system during flight. A hydraulic pressure transducer and the existing hydraulic low pressure switch was utilized for this purpose. A low pressure indication (say, under 500 psi) indicates a loss in hydraulic system pressure; i.e., a defective hydraulic pump or loss of fluid. A high pressure indication (say, exceeding 1150 psi) indicates a hydraulic pump failure with the relief valve maintaining maximum hydraulic pressure at 1150 psi; i.e., a failure of the pressure regulator which is an integral part of the hydraulic pump.

4.9.4.2 Hydraulic Pump Bypass Flow and Δ T

The pump by-pass flow back to the hydraulic reservoir is the leakage flow from the pressure regulating portion of the hydraulic pump. This flow gradually increases as the accumulated time on the pump increases. If this by-pass flow and by-pass temperature (i.e., difference between hydraulic oil temperature in the pressure manifold and hydraulic oil temperature in the by-pass line) is monitored, the condition of the hydraulic pump is known at any time. Also, by accumulating trend data of these parameters, the expected life of the hydraulic pump can be predicted. Further, an excessive increase in hydraulic

4.9.4.2 continued

oil temperature can mean internal leakage somewhere in the system or a low fluid level in the hydraulic reservoir. The above parameters are monitored in the UH-1 via a flowmeter in the hydraulic pump by-pass line and two thermocouples located with one thermocouple in the system pressure manifold and one thermocouple in the hydraulic pump by-pass line.

4.9.4.3 Hydraulic Filter ΔP

A hydraulic filter is installed in the main pressure line downstream of the hydraulic pump. The pressure drop across this filter was monitored by a ΔP transducer and examined for a high limit. A high value indicates a dirty filter for which the system will initiate maintenance instructions. A clogged filter is an indication of oil contamination or an incipient failure of a component in the hydraulic system.

4.9.5 Remaining Parameters Monitored in Aircraft

The parameters discussed below are monitored by various sensors located throughout the aircraft.

4.9.5.1 Total Pressure

Total pressure is monitored by sensing airspeed pitot tube pressure with a pressure transducer. This parameter is monitored for two reasons: (1) it is an important parameter used in the Gas Path Engine Analysis technique; (2) it is also used to monitor airspeed.

4.9.5.2 Static Pressure

Static pressure is monitored by sensing pressure in the aircraft instrument static pressure line utilizing a pressure transducer. This parameter is monitored for two reasons: (1) static pressure is also used to monitor aircraft altitude; (2) total and static pressure are definitive of airspeed.

4.9.5.3 Fuel Pressure

Aircraft fuel system pressure is a parameter that was already monitored in the UH-1 aircraft. This parameter was utilized in the UH-1 AIDAPS system to determine if fuel pressure at the fuel control input was sufficient for proper operation of the fuel control. When fuel pressure drops below 4 psi, a malfunction is present and the proper diagnostic message was displayed.

4.9.5.4 Left and Right Fuel Boost Pump Switch

These discrete signals are already present in the UH-1 fuel system. They are utilized in AIDAPS to determine if a fuel boost pump is not operating and also to determine which boost pump is malfunctioning.

4.9.5.5 Inlet Air Filter ΔP Switch

A ΔP switch is placed across the inlet air screen to determine if the airflow through the screen is obstructed by foreign debris. When the differential pressure reaches a dangerous level, the switch is activated and the proper diagnostic message is displayed.

4.9.5.6 28 VDC, 115 VAC Essential Bus Voltage; 26 VAC Instrument Bus Voltage

The 28 VDC and 115 VAC essential bus voltages along with the 26 VAC instrument bus voltage are monitored to determine if malfunctions may be present in the various electrical systems. Also, by trending the various bus voltages, incipient failures of critical components may be prevented; i.e., a bus voltage supply may be lowering over a fixed time period. A diagnostic message to perform more detailed manual checks can be displayed.

4.9.5.7 Hi/Lo RPM Warning Light and Audio

The Hi/Lo RPM Warning Light and Audio are signals already present in the UH-1 aircraft. The AIDAPS system utilized these existing signals to perform an operational check on the Hi/Lo RPM Warning System.

4.9.5.8 Overspeed Governor Switch Position

The Overspeed Governor switch is already present in the UH-1 aircraft. The AIDAPS system utilized this signal to determine whether the engine is in a manual or automatic speed control mode.

4.9.5.9 Collective Pitch Position

Collective pitch position is an indirect measurement of the angle of the main rotor blades. This parameter was originally chosen to correlate the rotor blade angle with specific power settings. It was later decided that this parameter was not required in the system.

4.9.5.10 Vibration Transducers and Parameters Monitored

Two major sources of vibration in each of the UH-1 power train components are bearings and gears. Velocity type transducers were chosen to measure the bearing vibration because these transducers provide more sensitivity at the frequencies and amplitudes expected. Accelerometers were chosen to measure the mesh and sideband frequencies generated by the gears because of the higher frequencies and amplitudes involved.

The components that were chosen for monitoring were parts within the power train components with the highest rate of failure as specified in Army maintenance records contained in Bell Report No. 299-099-520. The primary consideration in choosing mounting location for the transducers was to mount the transducers as close as possible to the part whose failure detection was desired. Brackets were designed and fabricated of AMS 5616 steel to make use of existing bolts or studs on the individual components for ease of installation. The locations of the vibration parameters monitored and the sensors used are listed in Table 4-1.

4.10 Sensor Selection

Sensors were selected for the UH-1 AIDAPS Test Bed Program after careful study of the recommendations made by Bell Helicopter Company and AVCO Lycoming Division (ALD) in Report No. 299-099-521 - Automatic Inspection Diagnostic and Prognostic System (AIDAPS) Test Bed Program - Task II.

In general, the vast majority of the parameters which were chosen for monitoring were recommended in the referenced report. All of the parameters recommended in the Bell Report were reviewed with regard to the objectives of the Test Bed Program. After this review, some Bell recommended parameters were eliminated in an attempt to accomplish the program objectives in the most cost effective manner since additional parameters were required to enable the Gas Path Analysis technique to be performed as a diagnostic and prognostic tool.

Sensors selected fall into the following major categories:

1. Those which were existing in normal UH-1 installations which could be monitored in parallel with existing gages, alarm lights, etc.
2. Those which were selected by ALD based on experience with instrumentation of the T53 engine used in the UH-1 aircraft.
3. Those which were selected by Hamilton Standard based on extensive experience in the field of instrumentation.

Sensors were calibrated at the outset of the program and this calibration data was used directly in the engineering unit conversion of all data taken during the course of the program. The original intent was to recalibrate all sensors periodically during the course of the program. However, all of the necessary facilities to perform this calibration were not available at Corpus Christi. Furthermore, schedule commitments did not allow sensors to be transported back and forth to HS to perform the calibration. Sensors were

4.10 Continued

thus recalibrated at the conclusion of Phase D and the ensuing review indicated that the majority of the sensors remained within the predicted accuracy. Some problems were experienced with sensors during the program, but they can be attributed mainly to the constant assembly and disassembly which was required because of the numerous mechanical components which were tested.

A tabulation of sensor characteristics is shown in Table 4.1.

4.11 Documentation of Hardware Problems

Table 4.2 of this report is a chronological listing of all hardware problems that were encountered during Phase D of the Test Bed Program. The problems are typical of those encountered in most R & D programs.

Since this program is a Research and Development program, the hardware used had to have flexibility for changes; therefore, production hardware was not used. In order to have flexibility for change, some printed circuit

and the electronics chassis had to be "hard wired". This makes the electronic units more vulnerable to malfunctions and wiring problems. Also, due to the numerous LRU's changes, the electronic hardware sensors, and harnesses were removed and replaced many times. This caused accelerated wear of the harnesses and subjected the electronics and sensors to excessive handling, thereby making them vulnerable to damage.

SECTION 5

DATA ACQUISITION AND PROCESSING

5.0 DATA ACQUISITION AND PROCESSING

5.1 Introduction

An understanding of the data acquisition and processing methods is clarified by segregating the two categories of information to be acquired: digital data and vibrational data. A discussion of the digital data acquisition is presented in Paragraph 5.2 and the general processing techniques are discussed in Paragraph 5.3. The remainder of this paragraph presents additional explanatory information on the AIDAPS system.

The primary component in the digital data acquisition process is the Main Electronics Unit (MEU) which was installed in each helicopter. The MEU contains the necessary arithmetic and logic units required to convert the sensor data into meaningful measures of the various signals. A detailed discussion of this unit is presented in Section 4.2. The MEU is capable of performing all limit and logic tests which are required by a field AIDAPS installation.

The Maintenance Action Annunicator Panel (MAAP) was provided to display the results of the airborne diagnostic process. The mechanical details of this component are presented in Paragraph 4.4 and the software logic which activates the various indicators is presented in Paragraph 5.3.3.2. The Data Entry Panel (DEP) was used to enter documentary data such as engine serial number and to examine the data during the flight. The mechanics of this component are discussed in Paragraph 4.3.

The AIDAPS Test Bed Program was, by its very nature, a development program which implies the use of several approaches in examining the actual flight data. Several ancillary components were then included to facilitate this

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5.2 Digital Data Acquisition

The major functions of the MEU in acquiring the digital data are depicted in Figure 5-1. The AIDAPS system monitored the outputs of 46 individual sensors, 26 switch closures, and 10 self health parameters indicative of the MEU operation. These signals or parameters are introduced to the MEU, amplified as required, and converted to a digital representation for storage via software control. The program logic or software then stores the data, prepares it for output to the various ancillary devices, and conducts an analysis of the data.

5.2.1 Input Frame Composition and Scan Rate

The actual number of parameters to be monitored is not restricted to those values presented above. The total number of samples can be, and actually were, much greater than 46 in a basic sampling frame. This sampling frame was established as two seconds during an early phase of AIDAPS by the recorder requirements and the number of parameter values to be acquired. The basic sampling frame is further divided by clock pulses into segments of ten millisecond duration. Thus a maximum of 200 parameters could be interrogated in the two second frame if each parameter is addressed only once per frame.

Several different sampling rates are used in the actual AIDAPS program which results in more than one parameter value for each sensor within a specific frame. The actual sampling rates are established for slowly changing parameters (28 parameters once every two seconds), moderately changing parameters (17 parameters once every second), and fast parameters (17 parameters once every half second). An additional five segments are reserved for documentation

5.2.1 Continued

information from the FDEP. Thus, a total of 135 of the available 200 segments are presently utilized.

Limited input data compression is accomplished in the electronic interface for the switch closures. The 26 closures can each be represented by a 1 or 0 and each switch need not be assigned to a separate word. The switch information is actually packed into three data words and logic is provided in the software to isolate and extract the individual switch state.

5.2.2 Analysis and Data Displays

Several forms of output are provided for the Test Bed Program AIDAPS. Current display of any parameter value is provided by the FDEP. The FDEP displays the parameter number requested by the user and the voltage equivalent of that parameter. Documentary data such as aircraft and/or engine number may also be entered into the computer via the FDEP. The MAAP provides 3 incremental counters and 64 message indicators. The 3 counters document the flight duration, the number of recorded frames, and the number of engine steady state frames.

The number of MAAP messages was determined by reviewing the initial diagnostic diagrams, assignment of tentative messages, and then providing some growth capability. This procedure resulted in 64 indicators which were activated on demand at the end of the flight to indicate the maintenance action which would be required.

The development of the logic necessary to activate the MAAP indicators involved one noteworthy step. The initial flights indicated that many messages were being activated. Subsequent investigations revealed that spurious fluctuations of various parameters had occurred due to normal aircraft

5.2.2 Continued

transients resulting in the generation of erroneous messages. An interim change was initiated to separate the transient from the true maintenance action requirement. This modification involved counting the total number of incidents for each MAAP indicator, comparing this number of occurrences with the number of flight frames, and then activating only those indicators which had been encountered for a significant portion of the flight. The initial selection of the number of incidents required to activate any one message indicators was 25 percent of the total flight. This percentage is not fixed and subsequent analysis has indicated that a value of 10% could be satisfactorily employed. Analysis also indicated that certain logic groups are of a sufficient degree of severity to warrant a message activation for any single limit exceedance. One example of a critical diagnostic group involves the rotor overspeed and underspeed alarms. Improper operation of this warning system would have a significant impact on the aircraft safety in a deployed situation. A relatively minor modification is available to incorporate both of the message activating approaches into the MEU.

5.2.3 Output Frame Composition

The tape recorder is a seven track digital recorder. This means that 7 bits of data (1's or 0's) are recorded across the tape and the tape is incremented a fixed step between bytes (one byte is seven bits). One bit is used as a recording check (parity) and 6 bits per byte are used for data. An actual MEU data word consists of 16 bits but only 12 of these bits are used for each parameter value. Thus, two tape bytes are equivalent to each parameter and the program logic locates the proper word half and supplies it to the recorder in a timely manner.

5.2.3 Continued

The AIDAPS output frame to the recorder includes the 200 words of the input frame plus 37 documentary words for a total of 237 words or 474 bytes. The 37 documentary words include frame identifiers, counters, time, and test information. The recording rate is hardware clocked at 300 bytes per second and a total time of 1.580 seconds is required for each frame. This allows 420 milliseconds between frames for blank tape which is used as an inter record gap (IRG). The IRG also serves as a logical frame separation.

5.2.4 Conditions for Engine Analysis

The proper analysis of a complex system such as a gas turbine engine requires that the effect of random variables on the analysis be minimized as much as possible. This requirement was recognized in establishing ground rules for the AIDAPS analysis and led to the imposition of three requirements for acceptability of the engine data.

1. The engine must not be analyzed in a transient operating condition.
2. Care must be exercised so that non-linear engine characteristics will not generate false data.
3. The effect of engine to engine variations on the analysis must be examined and minimized if significant.

These considerations are discussed in some detail in subsequent paragraphs.

5.2.4.1 Engine Steady State Criteria

Extensive background in the development of turbine engine controls has demonstrated that the transient operation of an engine can significantly effect the diagnostic results. For instance, the occurrence of a high

5.2.4.1 Continued

steady state temperature would be indiciative of a potential engine degradation but that some variation which occurs during a transient would only be indicative of the energy available to accelerate the engine. Some method of minimizing the effect of transient performance was then required in the AIDAPS system. Attempts to include the transient characteristics would have resulted in an extremely complex engine model which would not have been economical. Tests were then established to isolate only the steady state conditions and then reviewed for validity.

The initial steady-state criteria was established based on known engines parameter characteristics such as operating range and response time in conjunction with the sampled data accuracy and the sampled frequency. These criteria were:

1. The gas turbine uncorrected speed (N_1) readings within a 2 second recording frame must not differ by more than 100 RPM.
2. The uncorrected main rotor speed (N_R) single frame averaged value, must be greater than the minimum normal flight limit of 295 RPM specified by Bell Helicopter Co.
3. The corrected shaft horsepower (SHP_c) must be greater than 150 HP to eliminate autorotation conditions.
4. The corrected averaged value of exhaust gas temperature ($TT9_c$) between adjacent recording frames must not exceed 2.5°R.

The steady-state criteria optimization was performed using the 5-13-71 (AM) flight of aircraft No. 17223. This flight was selected because the flight plan had included extremes of aircraft operation (i.e., high rate climbs and descents) in addition to normal flight patterns used to obtain

.2.4.1 Continued

gas path steady state data. The resulting gas path data obtained from this flight and processed on the HS IBM 370 computer is presented in computer plot form in Figures 5-2 through 5-6. In these plots of compressor discharge pressure, compressor discharge temperature, fuel flow, shaft horsepower, and exhaust gas temperature, minimum, maximum, and averaged values are plotted at the mid-point of ten (10) N_{1c} speed windows.

From these results, it was concluded that the percentage spread between the minimum and maximum values (at constant N_{1c}) of W_F, SHP, and T_{T9} was unacceptable. A detailed evaluation of the flight data showed that the T_{T9} data spread was being caused by the engine's slow thermal response characteristics (i.e., for large and/or rapid changes in collective pitch setting transient T_{T9} data was getting into the gas data).

The final steady state criteria accrued represents the optimal compromise between the desired steady state criteria and the number of gas path data points necessary to perform a good gas path analysis. This criteria is presented below:

1. The gas turbine uncorrected speed (N_1) readings within a 2 second recording frame must not differ by more than 200 RPM.
2. The gas turbine corrected speed (N_{1c}), single frame averaged value, cannot differ from an adjacent frame averaged value by more than 100 RPM.
3. The uncorrected main rotor speed (N_R), single frame averaged value, must be greater than the minimum normal flight limit of 295 RPM specified by Bell Helicopter Co.

5.2.4.1 Continued

4. The corrected shaft horsepower (SHP_c) must be greater than 150 HP.

5. The corrected exhaust gas temperature (T_{T9C}), single frame averaged value, cannot vary by more than $\pm 2.5°R$ in 60 seconds (or between frames if the time interval between frames exceeds 60 seconds).

Plots of the gas data measured parameters for the 5-13-71 (AM) flight, using this steady state test criteria, are shown in Figures 5-7 through 5-11. The table below shows the relative improvement in data scatter that was obtained. The scatter band percentages have been based on the average baseline curve for each measured parameter.

MAX. SCATTER BAND (%)

	Initial	Final
P_{S_3}	2.0	1.0
T_{T_3}	1.5	.75
T_{T_9}	4.0	.75
W_F	10.0	4.25
SHP	9.75	5.0

Studying Figures 5-7 through 5-11 shows that the averaged value at a particular speed window was shifted by as much as 1.0% on SHP_C between these two steady state criterions. This indicates that even a more stringent steady state criteria is desired. More stringent driterias were evaluated, but the reduced amount of steady state data was unacceptable for a gas path analysis.

The maximum scatter band percentages for W_F and SHP, although they appear large, are not excessive by statistical analysis standards. The 1 sigma standard deviations are 1.1 and 1.9%, respectively.

5.2.4.2 Engine Baseline Characteristics

The fundamental approach to engine analysis involves comparing the present engine condition with the performance of a good engine. This requires that a baseline characteristic be defined such that deviations from the baseline are then indicative of the engine health. The question then arises with regards to the resolution of the baseline curves. Typical T53-L13 engine characteristics were derived, and a study of this data indicated that an adequate definition could be obtained using four straight lines to approximate the curves. The usable power range was then defined by these four power windows and flight data collection was initiated in each of the windows. The power ranges for the windows are:

Window	Horsepower Range	Compressor Pressure Range P/δ_1
1	1100 to 1400	94 to 106
2	770 to 1100	82.5 to 94
3	470 to 770	71 to 82.5
4	250 to 470	59 to 71

A separate slope and intercept was established for each power window and the need for elaborate curve fitting algorithms was eliminated. The data was actually collected as a function of compressor pressure ratio as indicated in the above table. The validity of collecting data as a function of either pressure or horsepower is developed in Section 6. Compressor pressure was selected as the base parameter for the AIDAFS Test Bed Program because of the ease in measurement.

5.2.4.3 Engine Variability

The baseline characteristics of turbine engines have a strong impact on the credibility of the analysis. A significant engine to engine variation will generate false conclusions when compared with nominal characteristics. This variation for the UH-1 engines is quite large and, when combined with the conservative degraded parts selection, resulted in the need for individual ("customized") engine baselines. These baselines for the engines encountered during the AIDAPS program are presented in two sets of figures. The four flight test engines are presented in Figures 5-12 through 5-16, and the six final demonstration engines are presented in Figures 5-17 through 5-21.

A study of the baselines for 10 engines provides an interesting insight into the detection problem. These curves reflect a reasonable variation between engines. However, the engine to engine variation is quite significant in evaluating the health of the engine. The actual engine variations and resultant degree of uncertainty of the implicit or calculated parameters are tabulated in Figure 5-22. These variations of the implicit parameters are quite large and are unacceptable for a reasonable diagnostic system. Thus the need for accurate individual baseline characteristics is established in order to minimize the degree of uncertainity of the results and to obtain proper and timely diagnostics.

5.2.5 Adaptive Recording Control

Adaptive recording was provided as a means of major data compression during a flight. This technique was employed as a method of obtaining critical data without accumulating an unwieldly mass of background information. The following tests were devised to initiate recording:

5.2.5 Continued

1. Record once each minute; or

2. Record whenever 10 exceedances of the total of 135 limits were encountered in a frame; or

3. Record all valid gas path frames as defined by the steady state tests of paragraph 5.2.4 and record typical frames as steady state engine conditions are being approached.

These tests then compressed the large amount of information available on a flight into a meaningful flight history.

5.2.5.1 Periodic Recording Mode

The ability to record at least once every minute provided the means of monitoring individual sensor operation. These records were then examined for proper parameter values as a check on the system.

5.2.5.2 Limit Exceedance

The recording of every frame for which 10 out of 135 limits were encountered served to document critical operating conditions. The value of 10 limits was an arbitrary selection and can be adjusted either up or down in a deployed system.

5.2.5.3 Steady State Conditions

All data frames which satisfy the engine steady state conditions are recorded for subsequent ground analysis. This step was provided in order to properly document the flight for additional software review and/or modification.

The recording of typical frames as steady state conditions were being approached was provided in order to aid in documenting the flight history. These frames then provided the data base used in establishing the proper engine steady state tests.

5.2.6 Flight Profile Definition

An important requirement of any development program is to obtain definitive data at controlled operating conditions. A desired flight profile for the AIDAPS missions was established to satisfy this objective. This profile, as presented in Figure 5-23 defines the desired speed settings and the time at each condition as well as pertinent data which should be noted. The exact sequence is relatively unimportant in this profile since the MEU logic is established to isolate the power conditions without regard to time. The factors which were important include maintaining constant airspeed and constant altitude, at each N_{1c}. N_{1c} is calculated as follows: $\% N_{1c} = \% N_1$ Indicated + N_1 T_1 from curve. It should be noted that this desired profile was only generated as a guideline for the Test Bed Program. Valid data can be expected from normal missions, and the need for special diagnostic flights in a deployed installation is not indicated.

5.3 Digital Data Software Discussions

The primary data processing in a deployed AIDAPS will occur in the airborne processor, in this case the MEU. Ground processing may be desirable for some commercial applications but is not a feasible requirement in AIDAPS. However, ground processing was employed in the Test Bed Program in order to clarify the test bed developments. This section then discusses the typical software programs which enabled the development of a successful AIDAPS concept.

5.3.1 AIDAPS Airborne Software Discussion

The general requirements for the airborne program were discussed in paragraph 5.2. The following steps were included in the airborne program and are typical of those logic groups which would be required in a deployed situation.

1. Initialization Test (once per application of power)
2. Parameter Input Sequencing (once, parameter each 10 m sec)
3. Parameter Output Sequencing (one frame/2 sec)
4. Mechanical Limit and Logic Testing (as required up to once every 10 msec outputted at end of flight)
5. Gas Path Data Collection (once per frame)
6. Gas Path Analysis (once per flight)
7. Maintenance Action Display (once at end of flight)

A flow diagram of a typical AIDAPS airborne program is presented in Figure 5-24 to clarify this discussion.

5.3.1.1 Initialization

The first locations of memory are reserved for troubleshooting. These locations are exercised only during a power interrupt or as the result of a program logic error. The latter feature is useful in checking new programs but is not encountered during normal flights. The logic then functions when power is turned on to apply proper tension to the magnetic tape, initialize counters and subroutine entries, and clear the FDEP. The logic then synchronizes on the 10 millisecond pulse (ref. paragraph 5.2.1) and decides what type of action is required. Possible actions at this stage include recording one byte if the frame is not complete, initialization for the next frame of either inputs or analysis if the flight is finished, or program termination at the end of the flight and data analysis. Definition of a completed flight is accomplished via a pushbutton on the FDEP which changes the data collection routines to data analysis routine.

5.3.1.2 Parameter Input and Data Process Sequence

The proper input subroutine for each parameter is selected, the data is brought into the MEU and stored, and the digitization process for the next parameter is initiated. The logic flow then divides into two branches. Gas path analysis is done only during those times that recording is not required, otherwise two more bytes of data are recorded. Finally, the mechanical limit and logic tests are run on each parameter as received. The MAAP display is energized at the end of flight test as indicated by the status of the DEP switch based on number of logic faults. The program then cycles back to the ten millisecond pulse logic and repeats this cycle.

5.3.1.2 Continued

The mechanical limit and logic tests are basically small, table driven routines. This means that a few general logic routines are used for all variables. The number of the parameters to be tested, the type of tests, the limits, the lamp number for the MAAP, and any special features are condensed into small tables. These values are extracted and expanded at the proper time to control the logic routines. This approach was utilized in order to minimize the memory requirements. A more complete discussion of the actual mechanical logic is not required for a basic understanding of the airborne program. The complete logic is documented in Section 6.1.

5.3.1.3 Data Entry Panel Input Routine

An interesting input subroutine was devised for the FDEP. Data may be displayed by activating a DISPLAY button and entering a parameter number from 1 to 200. Panel data may be stored for recording by activating an ENTER button or recording may be initiated using the EVENT button. Finally, recording will be initiated if a given number of limit exceedances have been detected. The panel routine returns directly to the 10 millisecond pulse logic because the subroutine is entered only when recording is not required.

5.3.1.4 Gas Path Calculations & Logic

Detailed gas path derivations and discussions are beyond the scope of this section and are presented in Section 6.2. A general review includes data analysis during the flight and logic at the end of the flight. The data analysis includes steady state tests, tests for the proper power condition (window) and evaluation of the parameter variation from a pre-established baseline. The logic includes noise elimination, calculation of the implicit

5.3.1.4 Continued

engine variables from the explicit or measured parameters, and the detailed gas path logic. Finally, the proper MAAP messages are activated and the program returns to the 10 millisecond pulse logic from either the collection or analysis sections.

5.3.2 AIDAPS Ground Data Analysis

The ground data analysis consists of two segments; on site data review and remote analysis.

5.3.2.1 On-Site Data Review

The on-site data review is accomplished in the Mobile Office and utilizes the DDP 116 digital computer. Data is read from the 7 track tape one frame at a time and stored in the computer. The parameter values may then be printed in engineering units on the ASR-33 teletypewriter. A typical frame is shown in Figure 5-25. A brief parameter functional check may then be conducted to isolate and correct for any sensor malfunctions in a timely manner. The 7 track tape is then sent to the main Hamilton Standard plant for a more critical analysis.

5.3.2.2 Remote Data Analysis

The remote data analysis is accomplished via an IBM 370 Model 155 Digital Computer. A general flow diagram of this process is shown in Figure 5-26 The 7 track tape is first edited to eliminate noise and supurious data and two 9 track IBM compatible, image tapes are generated. These tapes contain the good airborne data with the digitized data converted back to volts. The second or safety tape is obtained in order to maintain data security. The airborne sensors and electronics have been periodically calibrated and this

5.3.2.2 Continued

data is supplied to the computer via punched card input. The main functions of the computer program are summarized in the central block of Figure 5-26 and will be discussed in more detail in subsequent paragraphs. The output of the program is the printed data, gas path analysis plots, and data cards for the TREND program.

5.3.2.3 <u>General IBM 370 Program Discussion</u>

The principal function of the computer program is to reduce the data from the flight into an effective summary. Several major steps are outlined in Figure 5-26 and discussed in the remaining paragraphs of this section. The first step is to convert the tape data from airborne computer volts back to the proper engineering units. The individual frame data may be printed as specified by the programmer and sample frames are shown in Figures 5-27 through 5-29. Each page represents one 2-second burst of data and includes the frame number, time from the start of flight, and other documentary information. The gas path parameters are printed on the left side of the page and depict the four values in a frame under the parameter title. The twelve values of the exhaust gas temperature ring are printed in a column and generally separate the internal engine parameters from the external parameters (ambient temperature and pressures) and mechanical parameters. The mechanical parameters include the titles and readings (1, 2, or 4 values) immediately to the right of the title.

The center of the frame page is devoted to switch closure information. Each switch is interrogated twice in a 2 second frame (slot 1 and slot 2) and may be either a 1 or a 0 to designate an open or closed position. The third

5.3.2.3 Continued

column after the switch identifier is the malfunction code which is also either a 1 or 0. A malfunction would be detected in the switch parameter if the slot values agree with the malfunction code. Normal operation is obtained when the slot readings are the complement of the malfunction code.

The final portion of the engineering unit page is a listing of average frame data and the gas path data as corrected for ambient pressure and temperature variations.

The second step in the computer program is the mechanical logic tests as discussed in Section 5.3.4. The results of these tests are summarized and printed at the end of the tape analysis as a diagnostic message. The average values of pertinent parameters are also stored for printing as TREND data.

Gas Path Analysis is the remaining portion of the computer program and consists of those steps discussed in Section 5.3.5 including steady state tests, baseline comparisons, implicit variable evaluation, and diagnostic logic. The outputs of this step are the diagnostic message summaries, tabular gas path data for detailed review, and plots of the critical engine parameters. These plots are especially useful in appreciating the data because the results of an entire flight can be quickly observed. One example of this usefulness is discussed in paragraph 5.2.4.2 with regards to the engine steady state test definition. The parameter plots revealed a large and unacceptable variation which was subsequently traced to an insufficient stability test.

5.3.2.4 <u>Engineering Review</u>

The final step in the ground processing is the engineering review and analysis of the flight data. The complete data printout must first be reviewed

5.3.2.4 Continued

for rational operation of all sensors. Current data and diagnostic information is then analyzed to confirm that the analytical technique is correct. Program revisions have been made as required to improve the analysis. These changes were incorporated into the flight software where required. Finally, the information is summarized and filed for future reference.

5.3.3 <u>Detailed Program Discussions</u>

The AIDAPS Test Bed Program encompassed several disciplines which can be discussed individually. These areas include

1. Mechanical diagnostics (Section 5.3.3)
2. Gas Path diagnostics (Section 5.3.4)
3. Prognostics or Trends (Section 5.3.5)

The individual logic routines are discussed in the appropriate sections as indicated above.

5.3.3.1 <u>Mechanical Diagnositcs</u>

The basic mechanical diagnostics approach which was derived from several sources includes establishing reasonable limits for the important parameters, limit exceedance tests on these parameters, and then confirming the results of the limit test by the investigation of related parameters. Thus redundant checks will establish whether a simple inspection or a full replacement is required.

The mechanical logic definition had two other goals. First, the logic must be adaptable to the flight mode. Complicated cross checks, logic tests, and limit interdependencies for each parameter would rapidly use up the available computer memory and would result in an impractical system. Second,

5.3.3.1 Continued

the amount of installation and associated hardware implications must be within the scope of the basic AIDAPS program. These goals were then employed in reducing the number of potential candidate parameters to a reasonable definition of the critical components.

The assistance of the Bell Helicopter and Avco Lycoming were actively sought in fulfilling the above stated goals. The government furnished data which included the Bell reports * and other manuals which were helpful in defining the initial parameter limits and dependencies. The diagnostic logic tree, which is discussed in detail in Section 5.3.3.2, was then established. A review of this logic by Bell representatives was valuable in refining the diagnostics for the airframe area of concern. The Lycoming data package assisted in defining the engine mechanical relationships.

Some aspects of both the Lycoming and Bell diagnostics had to be deleted to arrive at a practical system for the AIDAPS goals. The compressor operation was deleted as being redundant with Gas Path Analysis. Similarily, hot starts and low cycle fatigue are very long term effects which would not be encountered in the relatively short AIDAPS program exposure. Thus, both data packages were edited in order to obtain a reasonable demonstration approach to airborne diagnostics.

The finalized limit values are presented for each parameter in Table 5.1. Certain limit revisions were required as additional information on the helicopter operation became available. Examples of the limit changes include the AC Essential Bus Voltage and the Hydraulic Pump Temperature Rise and Bypass Flow. The initial voltage band was increased from 5 to 10 volts to agree with

*Bell Helicopter Technical Reports No. 299-099-520, dated 1/30/70 and No. 299-099-521, dated 2/12/70.

5.3.3.1 Continued

the latest maintenance instructions. The hydraulic temperature rise was increased from 30 to 50 degrees and the leakage flow from 0.13 to 0.45 gpm to reflect the normal pump operation as compared with the operation of a new pump. The lower limits were determined to be acceptable only for a new pump in a test stand checkout and would have resulted in continual pump replacement without cause.

Recent flights have indicated the need to re-examine the lower limit of the DC Essential Bus. Additional aircraft power demands have resulted in this voltage being $\frac{1}{4}$ to $\frac{1}{2}$ volt below the lower limit with satisfactory operation. This indicates that the lower limit could be reduced from 26 to 25 volts.

In summary, a reasonable effort was expended in devising, checking, revising and implementing a workable approach to mechanical diagnostics. The result of this effort is the detailed logic tree as presented in Section 5.3.3.2. This diagnostic logic was incorporated into both the airborne and ground software programs and the results are discussed in Sections 6.0 and 7.0.

5.3.3.2 Detailed Mechanical Diagnostics

The mechanical diagnostic logic which are used in the airborne and ground based programs are identical. The actual logical flow diagrams and maintenance action messages are presented in Figures 5-30 through 5-39 and a detailed discussion will be presented as the final portion of this section. A brief overview of the Hamilton Standard approach is now presented to improve the understanding of later discussions.

5.1.6.7 Continued

The mechanical messages, M1 through M47, are listed in Figure 5-39 and are referenced by the logic diagrams of Figures 5-30 through 5-38. The first 17 messages pertain to the engine and are the result of a series of limit exceedance tests. Each test is shown in a block and the logic flow follows the "yes" or "no" branch depending on the results of the test. For example, message M2 will be activated only if the following conditions exist:

Bearing #2 pressure (B2P) must exceed 8 psi and either bearing #2 temperature rise must exceed 220°F or bearing #2 must have chip indication.

The logic for message M3 can be written as:

Engine Oil Temperature greater than 250°F

and

either bearing temperature rise greater than 220°F

or

both bearing temperature rises greater than 220°F.

Review of the engine diagnostics shows that most messages require a double exceedance to activate or flag a message. A similar technique applies to the gear box logic for messages M18 through M21 in that a single exceedance require an inspection while two exceedances require a removal.

The logic for the hydraulic system (M22 through M25), the fuel system (M26 through M28), and the electrical system (M29 through M32) is direct and self evident. The transmission logic generates messages M33 through M41.

5.3.3.2 Continued

The logic for M65 through M68 was derived through discussions with Bell representatives and illustrates the degree of isolation which may be obtained. This block can detect a bad temperature switch (M65) or the oil cooler thermal valve (M66) or an obstructed oil cooler (M67) or poor fan operation (M68).

The final set of logic is directed towards the rotor speed warning system. The intent here is not to duplicate the present alarm system, although that could easily be accomplished, but to check for proper operation of the warning system. This logic generates messages if the audio or visual alarms are on incorrectly or are not activated when required.

5.3.3.3 <u>Comments on Lycoming Hot End Analysis</u>

The Bell Helicopter Company reports referenced in Section 5.3.3 included a procedure for estimating the effects of creep, low cycle fatigue, and thermal shock on the life of the engine turbines. These functions are also incorporated into an "Engine Usage Indicator" developed by the Avco Lycoming Engine Company. The applicability of this hardware component was considered in the initial data reviews but was subsequently deleted as being not applicable in the scope of the Test Bed Program. The reasons for this deletion are presented in the following paragraphs.

The basic hardware component developed by the Avco Lycoming Company consists primarily of analog function generation and integration circuits which in turn activate three mechanical counters. The functions implemented in this component are within the capabilities of the airborne MEU and the basic Lycoming unit was thus not required. However, the mechanical counters were purchased in anticipation of a software implementation of the hot end functions and diagnostics.

5.3.3.5 Continued

The Lycoming Hot End Analysis consists of three general tests as a function of exhaust gas temperature (EGT), gas producer speed (N_1), and the changes in each. These tests then estimate the following turbine conditions:

1. Creep;
2. Low Cycle Fatigue; and
3. Thermal Shock.

The three tests are each discussed in separate paragraphs. The various limits and units were defined by Lycoming and extracted from Appendix A of the referenced Bell reports.

Creep maintenance is requested when the creep rate exceeds 8 units/minute or when the total exposure exceeds 200,000 units. This creep rate would not occur until steady state conditions of 101% and 1600°R are encountered. These conditions were not experienced in the Test Bed Program. The total exposure of 200,000 units would require either very long operating times or significant engine deterioration, neither of which were encountered. For instance, steady state operation at 90% N_1 and 1600°R would necessitate over 1800 hours of operation to accumulate the total exposure of 200,000 units. Thus creep did not appear to be a major candidate for the Test Bed Program.

Maintenance as a function of low cycle fatigue is requested when its accumulated index exceeds 45,000. The index is a function of the changes of N_1 and EGT and a large number of hot starts would have been required to reach the exposure index threshold. The index for a speed change of 100% and an EGT change of 1700°R is only 11 units and 4000 starts of that severity would have been required to exercise the logic. Thus low cycle fatigue was not a prime diagnostic candidate for the Test Bed Program.

5.3.3.3 Continued

Maintenance as a result of thermal shock was recommended when its total index exceeded 30,000 units. This index is a linear function of the rate of change of EGT and has a slope of 6 units per 100 degrees/second. Thus approximately 5000 large accelerations would have been required to exercise this logic. This magnitude was also judged to be beyond the scope of the Test Bed Program.

In summary, the Hot End Analysis Technique as outlined in Appendix A of the Bell reports was reviewed in detail and was judged to be beyond the scope of the AIDAPS Test Bed Program. The basic approach is within the capability of the AIDAPS MDU and could be included in a pre-production system to resolve the remaining issues if so desired. The major unresolved issue is what exposure index should be used following the overhaul of an engine. This topic should be investigated by the engine manufacturer prior to the preproduction activity.

5.3.4 Gas Path Analysis Program

The derivation of the basic analytical technique and explanation of the fundamental principles involved is discussed in Section 6.2. This section discusses the application of the technique to the airborne system. The use of an IBM 370 computer as a development and verification tool for the Test Bed Program will also be documented although this computer is not required in a deployed situation.

5.3.4.1 Explicit and Implicit Parameters

Nine engine signals were measured for the AIDAPS Gas Path Analysis. These signals are ambient pressure and temperature, compressor discharge pressure and temperature, fuel flow, power turbine torque, exhaust gas temperature, and gas producer and power turbine speeds as discussed in Section 4.9. These measured parameters are then combined to form 6 explicit engine parameter ratios:

$$\delta_1 = P_T/14.7$$

$$\theta_1 = T_{T1}/519.$$

$$P_{S3C} = P_{S3}/\delta_1$$

$$T_{T3C} = T_{T3}/\theta_1$$

$$W_{FC} = W_F/\delta_1\theta_1^{0.5}$$

$$SHP_c = \theta_2 N_2 / \delta_1\sqrt{\theta_1}$$

$$N_{1c} = N_1/\sqrt{\theta_1}$$

$$T_{T9C} = T_{T9}/\theta_1$$

The Gas Path Analysis then used P_{S3C} as the independent variable to define the baseline values of T_{T3C}, W_{FC}, SHP_c, N_{1c}, and T_{T9C} so that the percent variations of each measured parameter ratio could be evaluated. These known or explicit parameters were then combined to calculate the status of the

5.3.4.1 Continued

implicit engine health as defined by the following six parameters:*

ΔW_a = compressor airflow;

$\Delta \eta_C$ = compressor efficiency;

ΔA_5 (or $N_1 A$) = gas producer turbine area;

$\Delta \eta_T$ = gas producer turbine efficiency;

$\Delta \eta_{PT}$ = power turbine efficiency; and

ΔT_{5C} = turbine inlet temperature.

$\Delta A_n = - \Delta \eta_{PT}$ = power turbine area

*NOTE: See Section 6.2.2 for instructions on evaluating implicit engine variables from the measured dependent engine values.

The diagnostic logic is then based on these implicit evaluations as discussed below:

5.3.4.2 Data Collection

A flow diagram of the complete gas path analysis is presented in Figure 5-40 and the diagnostic messages are tabulated in Table 5.2. The data is first converted to engineering units and then the 6 explicit engine parameter ratios are evaluated. The engine steady state tests as discussed in paragraph 5.2.4.2 are then conducted. The frame of data is ignored if the engine is not in steady state or in an acceptable power window (refer to paragraph 5.2.4.3 for window definition). If steady state tests are satisfied, values for T_{T3C}, N_{1C}, W_{FC}, SHP_C and T_{T9C} are calculated as a function of P_{S3C} and the individual variation of the actual data from the respective predefined baseline values is evaluated. These 5 variations, ΔT_{T3}, ΔN_{1C}, ΔW_{FC}, ΔSHP_C, and ΔT_{T9C} (obtained directly from measured signals) and thus explicit deltas are then investigated to see if they have exceeded reasonable tolerances. If any sensors are suspect, this step completes a data collection segment of the program and data analysis may commence.

5.3.4.3 Data Analysis

The data analysis consists of evaluating the health of the engine and identifying the required maintenance action. The variations of the 5 explicit variations are symptoms of the engine health. These five explicit deltas are then combined to identify the variations in six implicit engine parameters:

ΔW_a = compressor airflow;

$\Delta \eta_c$ = compressor efficiency;

ΔA_5 = gas producer turbine area;

$\Delta \eta_T$ = gas producer turbine efficiency;

ΔA_n = power turbine area; and

ΔT_{T5} = gas producer turbine inlet temperature.

The required maintenance action is then dependent upon these 6 implicit variations. The first series of diagnostic tests are directed at the gas producer turbine: Variations in nozzle area and turbine efficiency may result in either inspection or replacement of the turbine rotor or nozzles. It is also possible to distinguish between closed or coked or eroded nozzles.

The next test is brief and identifies power turbine problems. A more complete analysis and diagnosis would be obtained if the pressure and temperature between the two turbines could measured.

The next major series of tests involves compressor airflow and efficiency. A deterioration of either parameter will result in a warning or caution message. Sufficient degradation of both parameters yields the conclusion that the compressor rigging needs adjusting or that the compressor should be replaced.

5.3.4.3 Continued

The final series of tests contain two branches which are dependent on the results of the preceding analysis. Fuel flow is examined if no fault has been detected and turbine inlet temperature (ΔT_{T5}) is examined if a fault has been detected. A fuel flow deviation, with no other diagnostic, would reflect either a sensor malfunction or a deviation in the associated burner hardware. The operation of the fuel flow sensor is investigated during the preliminary analysis steps as previously discussed. The fuel flow deviation is then indicative of the health of the burner nozzles, liner, and diffuser. The turbine inlet temperature change is examined to define the need for an overtemperature inspection. The need for this inspection is based on an extrapolation of the data obtained during normal engine operation at moderate powers. If the turbine temperature is high at moderate powers it will probably also be high and exceed its fixed limits if the engine controls requested a transient high power condition. This high temperature diagnostic is thus inferred from normal flight conditions.

5.3.4.4 Diagnostic Limits

The diagnostic limits which are shown in Figure 5-40 as L1 through L8 are quite flexible and may be adjusted as desired in a production system. Fixed limits of 25% for L1, 2.5% for L2 through L6, and 5% for L7 and L8 were used in the Test Bed Program because of the conservative approach used in selecting degraded components. These limits would be much larger in a production system and may even vary with the particular engine. For instance, two new engines may exhibit an airflow difference of 10% as discussed in

5.3.4.4 Continued

paragraph 5.2.4.4. The diagnostic limit for the better engine would then be 10% higher than that imposed on the poorer engine. The Gas Path Analysis thus has sufficient resolution to isolate the multiple degradations commonly produced by normal wear and still incorporate normal engine to engine tolerances.

5.3.4.5 Ground Processing Diagnostic Investigations

The IBM 370 computer was used as a development tool in the Test Bed Program to examine alternate applications of the diagnostic technique. This flexibility was provided to improve the confidence in the final application as to the proper method of diagnosing engine health condition. Several alternate approaches were investigated as outlined below with the indicated results.

The individual frame data was examined to determine if an arithmetic average for each signal was correct or if a statistical "wild point" technique was required to improve the data credibility. This comparision exhibited little difference between the two sets of data and demonstrated that the simpler arithmetic average could be employed.

A diagnostic analysis based on an average of all valid frames in a given flight was compared to the same analysis of individual frames. This comparison confirmed that the analysis based on the average of all valid flight frames would yield proper results and tended to minimize the effect of odd frames. These odd frames were produced by the normally noisy environment and because an absolute steady state test cannot really be defined as outlined below. Analysis of individual frames could thus result in improper diagnostics.

5.3.4.5 Continued

The engine steady state tests as described in paragraph 5.2.4.2 were refined on the IBM 370 computer. Perfect steady state tests, or a test for absolute stability, would require that a condition be unchanged for long periods of time which is an impractical requirement. The combination of reasonable steady state tests and averaging an entire flight then tends to minimize the effect of individual frames of data which are collected during the approach to steady state.

A diagnostic analysis based on an average of all valid flight frames was compared with the analysis based on the average data in each power window. This comparison confirmed that multiple power windows are required to define the baseline characteristics and that an average set of variations for the entire flight would yield acceptable diagnostic results.

A comparison of the analytical results using baselines as a function of compressor pressure (P_{S3}/δ_1) and compressor speed ($N_1/\sqrt{\theta_1}$) indicated that the use of pressure yielded somewhat better results. The use of speed as the independent variable tended to amplify any inherent noise in the system through the action of the steeper baseline characteristics.

Thus the IBM 370 computer served as an effective development tool in refining the Gas Path Analysis technique. A commercial computer such as the IBM 370 is not a requirement for a production AIDAPS system because the diagnostics can be accomplished in the helicopter. However, such a computer may be desirable at the depot level if elaborate history files are to be maintained on an entire fleet of aircraft.

5.3.5 AIDAPS Trend Program Discussion

One objective of the AIDAPS Test Bed Program was to examine the helicopter engine operation for long term trends and to investigate the effectiveness of this tool in extending the time between engine overhaul (TBO). The two UH1-H helicopters which were used for the Test Bed Program had tail numbers 61011 and 17223. Helicopter 17223 was devoted to collecting data for the trend analysis and helicopter component changes were not introduced into this aircraft. The planned part changes of AC 61011 prevented data accumulation for any significant operating time with a fixed hardware configuration. Thus the flight data obtained from AC 17223 has been utilized exclusively for the trend analysis.

5.3.5.1 Trend Data Collection

The initial flight objectives were established in an attempt to collect data for every hour of flight. The accumulation of time on the aircraft was found to be very slow under this procedure and revisions were initiated in order to more rapidly accumulate aircraft time. These revisions included longer cross country flights and even some overnight flights. Data collection flights were then initiated every 10 to 20 hours of aircraft time. However, the total data exposure for this one aircraft was only 245 hours which is approximately 25% of the current TBO for the UH-1.

5.3.5.2 Trend Parameters

The parameters to be included in the trend data were limited to those parameters most indicative of the helicopter capability. These parameters and their respective trend limits are presented in Table 5.3. The gas path

5.3.5.2 Continued

parameter limits were derived by Hamilton Standard and reflect the present estimate of deterioration at which an overhaul should be initiated. It must be noted that a firm basis for these limits has not been established and that more long term data is required before a production-type limit can be defined. The mechanical trend limits were extracted from the Army furnished Bell Helicopter reports which are referenced in paragraph 5.3.4.

5.3.5.3 Trend Analysis

The trend analysis program utilized the data obtained from the IBM 370 programs in order to provide the capability of developing alternate analytical approaches. The current trend program consists of two distinct portions as discussed below:

1. Data Collection and Averaging; and
2. Data Analysis

5.3.5.3.1 Data Collection and Averaging

The data collection and averaging program functions to compact the valid information from a flight into the average value for each parameter of interest. Valid information for this program includes that data which has successfuly passed the steady state tests which were discussed in paragraph 5.2.4.2. The result of these steady state tests is depicted in Figure 5-41. The first sketch (I) illustrates a possible flight and includes both the steady state data and transient data. This transient data has been shown to produce erroneous results, and the elimination of it produces the data set shown in Sketch II. The previously discussed data collection then results in the

5.3.5.3.1 Continued

effective values of Sketch III. The initial portion of the flight is deleted because it is outside of the desired power windows. Thus the steady state tests also serve to compress the large amount of flight date into a meaningful measure of the helicopter operation.

The data collection program was established to isolate the valid data as a function of the four power windows discussed in Paragraph 5.2.4.3 in order to allow an examination of possible trends as a function of engine power conditions. Very little steady state data was obtained in either the high or low power window and most of the stable conditions occurred between 500 and 1000 horsepower. Subsequent discussions with Bell Helicopter representatives and analysis of the engine capabilities revealed that the UH-1H helicopter engine produced more power than required for the Test Bed Program loads. The extreme power ranges were only encountered during high velocity climbs and moderate velocity descents, respectively, and neither of these operating regimes are engine steady state conditions. The minimal data in the high and low power windows was then too scattered in time and quantity to be useful in the trend analysis.

Thus the result of the data collection program is a measure of the parameter values, the time of occurrence in the flight, and the number of valid data points in each flight for each power window which was encountered.

5.3.5.3.2 Analysis Program

The primary objective of the trend analysis program is to extend the current data and predict when an overhaul will be required on any component. Three questions are immediately raised in the prediction process.

5.3.5.3.2 Continued

1. What type of curve should be used to extrapolate the available data?

2. What is the proper point in history for the data?

3. Are the effects of unusal flights considered properly?

These questions are considered in the following paragraphs.

The best curve to be utilized in the prognosis should be obtained by monitoring the trend of many helicopters on a long term basis. This could be accomplished by establishing a data base during a pre-production program. However, this data was not available and alternate approaches were investigated. The current technique permits a first order approximation to most curves without an elaborate curve fitting algorithm. A series of straight lines are used to fit the data as shown by the examples in Figure 5-42. This figure assumes typical trend curves and shows how three straight lines may be used to approximate the data. The technique thus utilized the most recent data to predict short term trends (time span 1), the next eldest data to predict moderate trends (time span 2) and the oldest data to predict long term trends (time span 3). The time spans selected for the Test Bed Program were 10 and 100 hours for time spans land 2 respectively. Any data over 100 hours prior to the current aircraft time was used in time span 3.

The second question in prognostics concerns the proper point in history for the data. This question was of secondary importance in the moderate and long term trends but is important in establishing the short term trends of time span 1. The trend program utilized the aircraft age at the start of a flight (hours since overhaul) and the time into the flight at which the data was

5.3.5.3.2 Continued

encountered to define the actual time since overhaul at which the data was obtained. Thus data which is obtained during the first ½ hour of one 3 hour flight and during the final ½ hour of the next 3 hour flight was (and should be) analyzed using the actual time differential of 5 hours rather than the artifical time of 3 hours (the flight duration).

The third question involving the treatment of unusual flights was resolved by documenting the number of valid frames of data in each power window. A flight with few valid frames could then be considered as unusual if the data disagreed with the many valid frames which occurred in the flights before and after the unusual flight. Documenting the number of valid frames then allowed the effect of unusual flight on the normal trend to be minimized.

The final trend analysis program includes the previously described steps as presented in the flow diagram of Figure 5-43. The program first reads new data into span 1. Old data is shifted to the next span if the time spread in a span exceeds the desired time. The process is continued until all available data has been assigned to the proper time span and power window.

All data now exists as effective values for each parameter at four power conditions and 3 time spans. A linear equation is then determined for each parameter-power-time span set using the least squares fit approach. The data and trend lines are then plotted in order to graphically present developing trends. The derivation of the generic trend line equations is presented in Figure 5-44.

5.3.5.3.2 Continued

Finally, the trend lines are equated to the present malfunction limits and a total effective life for each parameter is defined. This life is the intersection of the trend line and the parameter limit, of Table 5.3. This time would then be useful in scheduling maintenance since it is the time at which a diagnostic would be encountered. The actual maintenance action to be performed would be obtained from the diagnostic logics of paragraph 5.3.4.2 for mechanical parameters and paragraph 5.3.5.2 for gas path parameters.

SECTION 6

SIGNATURE DEFINITION AND ANALYSIS OF FLIGHT DATA

6.0 SIGNATURE DEFINITION AND ANALYSIS OF FLIGHT DATA

6.1 Introduction

This section is primarily concerned with applying the data obtained during the flight phase of the Test Bed Program. The verification test results are documented in Section 7. Results of the Gas Path Analysis, and a derivation of the basic analytical technique, are presented in Section 6.2. Mechanical diagnostic logic results, which include the contributions of Avco Lycoming Engine Company and Bell Helicopter Company, are presented in Section 6.3. Section 6.4 documents the results of the vibration analysis.

6.2 The Gas Path Analytical Technique

6.2.1 A full appreciation of the Gas Path Analysis Technique requires that a basic understanding of the engine signals utilized be provided.

A schematic rendering of the AIDAPS engine is presented in Figure 6-1. The station designations are as indicated, and the general gas flow path is indicated by arrows. A brief discussion of the parameters which are utilized in the Gas Path Analysis is presented in the following paragraphs.

6.2.1.1 Engine inlet total pressure (P_T) was measured at the aircraft probe. This would be the same pressure that exists at station T and includes the effects of airspeed and altitude. Pressure P_T is a referencing parameter used to compensate for varying air density.

6.2.1.2 Compressor inlet temperature (T_{T1}) was measured at station 1. This temperature is a referencing parameter used to compensate for varying air density and to correct the internal engine temperatures for non-standard day conditions.

6.2.1.3 Compressor discharge pressure (P_{S3}) was measured at station 3. This pressure, in conjunction with P_T, is a measure of the pressure ratio being developed by the compressor and thus the general power potential of the engine. Compressor pressure ratio (P_{S3}/P_T) is used as the independent parameter in establishing engine baseline characteristics.

6.2.1.4 Compressor discharge temperature (T_{T3}) was measured at station 3. This temperature is important in defining the thermodynamic efficiency of the compressor and the gas producer turbine efficiency. Temperature T_{T3} exerts a secondary effect on the turbine inlet temperature and turbine area calculations.

6.2.1.5 Engine fuel flow (W_F) was measured at a low pressure point between the fuel tanks and the fuel control. Fuel flow is a measure of the energy available to the power plant and exerts a strong effect on compressor airflow and the turbine areas.

6.2.1.6 Exhaust gas temperature (T_{T9}) was measured at station 9 using the thermocouple ring which is inherent with this engine. This temperature is required to develop the thermodynamic efficiency relationships in the engine and has a major influence on all other calculations.

6.2.1.7 Gas producer speed (N_1) was measured at the available aircraft tachometer. This speed exerts a major influence on the compressor airflow evaluations.

6.2.1.8 Power turbine speed (N_2) was measured at the available aircraft tachometer. This speed is required to convert the measured torque to shaft horsepower.

6.2.1.9 Power turbine torque (θ_2) was measured using the torquemeter which is inherent with this engine. Torque and power turbine speed are used to recalculate

6.2.1.9 Continued

shaft horsepower which has a strong effect in defining the power turbine efficiency variations. Shaft horsepower has a secondary (but not negligible) effect on the other engine calculations.

6.2.2 Gas Path Analysis

A primary objective of the AIDAPS program is to evaluate the state of mechanical health of the airframe powerplant. In a general sense, the engine may be viewed as being comprised of accessory equipment, rotational mechanical equipment, and energy converting gas path elements. The accessory equipment include such elements as the fuel control, fuel pump, lubrication system, ignition system, hydraulic power system, and engine air bleed system. The rotational mechanical equipment includes the various gear boxes and transmission elements and the main engine bearings. The gas path elements include the compressor, the burner, the gas generator turbine and the free power turbine. Program treatment of the accessory and rotational equipment is dealt with in Section 6.3 - 6.5, and this section will confine itself to a theoretical consideration of the gas path elements. The actual implementation of the technique is presented in Section 5.3.

The parameters available to use as a basis for judgement of gas path relative health are various corrected temperatures, pressures, fuel flow, engine speed and output horsepower as outlined in paragraph 6.2.1. As a generality, differences in these parameters from their expected values can be used to inferentially determine which elements of the gas path have undergone distress, or departed from their initial or expected condition. The key word here is "inferentially"; it must be stressed that any parameter in itself is not directly indicative of faults in any particular element. For example, at any given speed a change in compressor discharge pressure does not necessarily mean there is a compressor fault. The change may be because of a combined compressor and turbine fault, or due to a turbine fault alone.

6.2.2 Continued

The primary independent variables of the gas path are the compressor air pumping capacity and efficiency, the burner efficiency, the gas generator turbine inlet nozzle area, inlet temperature and efficiency, and the power turbine inlet nozzle area and efficiency. These implicit parameters are not directly measurable but can be derived from various measurable explicit dependent variables.

Fundamentally, the parameters being measured are dependent variables whose absolute values depend on the absolute levels of all the primary independent engine variables. Concomitantly, changes in the primary but unmeasurable independent variables result in changes in the measurable dependent variable. Therefore, any diagnostic system which hopes to isolate gas path faults must be capable of interrelating changes in the dependent measurable parameters to changes in the independent unmeasurable parameters.

A flow chart showing the Hamilton Standard approach to this problem, used in the AIDAPS program is shown in Figure 6-2. Looking at this chart; since the approach is based on interrelating differential changes, the first task is to gather base line data in the useful power range on the parameters to be measured. On the T53L13 engine the corrected parameters available are $N_1/\sqrt{\theta_1}$, T_3/θ_1, $W_f/\delta_1\theta_1^y$, $HP/\delta_1\sqrt{\theta_1}$, T_9/θ_1 and P_3/δ_1. As shown in Figure 6-2, the first five parameters are stored as functions of P_3/δ_1.

If there were no engine-to-engine variation among new production line engines, the stored baselines for all engines could be a single set of nominal baselines. In point of fact, new engine variability is of the same order of magnitude as the deterioration changes being sought; and, therefore,

6.2.2 Continued

individual custom baselines of the particular engines in their "new" state must be stored. At any point in time after the engine has been used or abused (Installed Time > 0) measurements are repeated on the dependent variables and the differences calculated between their respective baseline values and present state values at the measured value of P_3/δ_1. These differences are used in an interrelationship matrix, explained subsequently, to determine what changes if any, and in whatever combination, have occurred in the basic independent parameters. The ability of Gas Path Analysis to sort out multiple faults is a key advantage over previous techniques such as a fault coefficient matrix. If there are any changes in any of the independent variables, these are put through a diagnostic logic routine which then ennunciates the probable causes of the fault or faults and the action to be taken.

From the preceeding, an obvious requirement for successful diagnostics is a means to interrelate changes in measured variables with basic engine faults. The most widely known prior technique in the industry relies on measuring changes in the dependent variables and comparing them with tables of pre-calculated expected deviations in these parameters (fault coefficients) for various possible engine faults to determine the statistically most probable single fault. The obvious shortcomings of this technique are that is is often statistically and thermodynamically possible to have multiple faults occurring wherein the effects of one fault will mask, cancel out or confuse the effects of the other fault on the chosen parameters, and that it is mathematically and thermodynamically possible for a set of multiple faults to have an undistinguishably similar effect on the chosen parameters as a totally unrelated single fault.

6.-.- Continued

The importance of this problem becomes obvious when it is realized that multiple faults must occur in a physical system such as a turbine engine. A properly designed system will experience a relatively uniform wear in all components (compressor, turbines, burners, etc.). A single failure of a component may occur due to unusual damage conditions but the general tendency will result in continual wear of all components. Thus the diagnostic system must isolate the multiple faults which will commonly occur. The new concept for Gas Path Analysis implied in Figure 6-2 has been previously developed by Hamilton Standard and applied to the AIDAPS program permits the simutaneous evaluation of all possible primary faults within the gas path.

A fundamental premise of the H.S. concept, as stated above, is that the measurable engine parameters are dependent variables, changes in which, at any given power and flight condition, are brought about by deviations in the fundamentally independent component performance parameters. Using the techniques found in "Gas Turbine Engine Parameter Interrelationships", a book published by Hamilton Standard, it can be shown that a general influence coefficient matrix may be written for the T53L13 engine defining the set of differential equations interrelating the various engine performance parameters. Looking at the major parameters of interest, it would be of the form shown in Table 6.1.

The means to use this general matrix to perform gas path analysis is as follows:

1. As previously mentioned, gather and store engines steady state baselines, over the operating range of interest, of $N_1/\sqrt{\theta_1}$, T_3/θ_1, $W_f/\delta_1\theta_1^{\,y}$,

6.2.2 Continued

$HP/\delta_1\sqrt{\theta_1}$ and T_9/θ_1 as functions of P_3/δ_1.

2) At any point in the engine's life, operate the engine in steady state, at any flight condition and at any arbitrary power condition within the baseline range, and remeasure the corrected parameters of step 1.

3) At the measured value of P_3/δ_1 compute

$$\frac{\partial N_1/\sqrt{\theta_1}}{N_1/\sqrt{\theta_1}} = \frac{(N_1/\sqrt{\theta_1})\text{ measured} - (N_1/\sqrt{\theta_1})\text{ base line}}{(N_1/\sqrt{\theta_1})\text{ base line}}$$

and similarly for the other corrected parameters.

4) Let the measured corrected quantity differences be called

$$\frac{\partial N_1/\sqrt{\theta_1}}{N_1/\sqrt{\theta_1}} = xa; \quad \frac{\partial T_3/\theta_1}{T_3/\theta_1} = xb; \quad \frac{\partial W_f/\delta_1\theta_1^{\,y}}{W_f/\delta_1\theta_1^{\,y}} = xc$$

$$\frac{\partial HP/\delta_1\sqrt{\theta_1}}{HP/\delta_1\sqrt{\theta_1}} = xd; \quad \frac{\partial T_9/\theta_1}{T_9/\theta_1} = xe$$

5) Extract from the general influence coefficient matrix its pertinent elements.

Per % Change In / Change IN	T_5/θ_1	$N_1/\sqrt{\theta_1}$ (=xa)	$\Gamma 1$	ηc	ηb	A_5	ηt	ηpt	
T_3/θ_1 =	.129	.746	.202	-.402	0	-.202	.028	0	=xb
P_3/δ_1 =	.397	2.299	.621	.087	0	-.621	.087	0	=0
$W_f/\delta_1\theta_1^{\,y}$ =	1.918	1.782	.482	.365	1	.518	-.072	0	=xc
$HP/\delta_1\sqrt{\theta_1}$ =	2.233	2.703	.731	1.778	0	.269	1.548	1.0	=xd
T_9/θ_1 =	.904	-.484	-.131	-.031	0	.131	-.261	-.153	=xe
An =	-1.097	1.849	.500	-1.347	0	.500	-1.450	0	

6.2.2 Continued

The numerical values shown are for a medium power condition ($P3/\delta_1$ = 78 psi, HPpt $\approx$ 750). The engine nonlinearities would introduce only minor second order differences in these coefficients at other power conditions.

6) This matrix represents a set of six differential equations in terms of six knowns and eight unknowns. To make this solveable, recognize the following:

a) Actual η b variations rarely occur and even then tend to be negligible. Burner problems are more likely to manifest themselves as variations in the downstream temperature profile and can be detected by the profile shown by the EGT rakes.

b) Power turbine nozzle (An) warping or burning and η pt variations will accompany each other. As a good first approximation let η pt change = -An change. Although this correspondence will not always be exact, at least the faults will be isolated to the proper component. More precise evaluation of the exact changes would demand measurement of at least one more parameter; for instance, P_7/δ_1, the inter-turbine pressure.

6.2.2 Continued

7) The rearranged matrix in its specifically used simplified form therefore is:

% Change In →	T_5/θ_1	Γ_1	η_c	A5	η_t	η_{pt}	
	.129	.202	-.402	-.202	.028	0	= xb - .746 x a
	.397	.621	.087	-.621	.087	0	= -2.299 x a
	1.918	.482	.365	.518	-.072	0	= xc - 1.782 x a
	2.233	.731	1.778	.269	1.548	1.00	= xd - 2.703 x a
	.904	-.131	-.031	.131	-.261	-.153	= xe + .484 x a
	-1.097	.500	-1.347	.500	-1.450	0	= -η pt - 1.849 x z

This now represents a solveable set of equations.

8) By matrix inversion, this matrix may be inverted to form the following simple set of equations, which are solveable by the basic operations of multiplication and addition:

% Change In	xa	xb	xc	xd	xe	
T_5/θ_1 =	0	.455	-.151	.151	.989	= ya
Γ_1 =	-3.700	-.007	1.340	-.340	-2.225	= yb
η_c =	0	-2.325	0	0	0	= yc
A_5 =	0	.230	1.266	-.219	-1.740	= yd
η_t =	0	1.936	.162	.175	-1.061	= ye
η_{pt} =	0	.063	-1.234	.699	1.529	= yf
An =	0	-.063	1.234	-.699	-1.529	= yg = -yf

The change in any unknown is simply the arithmetic sum of the multiples of its "b" coefficients times the measured knowns.

6.2.2 Continued

9) Thus, by measuring the changes in five corrected parameters at any arbitrary compressor discharge pressure, an evaluation has been made of the precise changes in all of the thermodynamic parameters of interest, and the physical parameters which contributed to these changes. Note that there is no need to measure actual air flow or high turbine inlet temperature - parameters which are traditionally very difficult to measure - yet the precise changes in these parameters from their baseline values have been evaluated. Although the change in turbine inlet temperature may be evaluated when operating below maximum engine conditions, this change will be a direct measure, ignoring very minor secondary effects, of how far over turbine inlet temperature the engine would be at high power lever settings where the fuel control would call for the engine to inferentially operate at the maximum turbine inlet temperature for a new fault-free engine.

Since the column headings of the original influence coefficient maxtrix (Table 6.1) are now completely defined, it follows that the change in any other internal gas path parameter may be evaluated, if this should prove of interest for any purpose.

Precise values of the matrix coefficients vary somewhat with engine operating condition. Therefore multiple sets are stored covering contiguous operating ranges or "windows", and the proper one chosen during operation to correspond with the P_3/δ_1 of the operating point.

6.2.2 Continued

10) To summarize:

a. At any arbitrary engine operating point, as defined by the measured compressor discharge pressure P_3/δ_1, changes have been measured in

$N_1/\sqrt{\theta_1} = xa$ $T_3/\theta_1 = xb$ $W_f/\delta_1\theta_1^{y} = x_c$ $HP/\delta_1\sqrt{\theta_1} = xd$

$T_9/\theta_1 = xe$

b. At the operating point the changes have been evaluated in

$T_5/\theta_1 = ya$ $\Gamma_1 = yb$ $\eta_c = yc$ $A_5 = yd$

$\eta_t = ye$ $\eta_{pt} = yf$ $An = yg$

and, if desired, any other thermodynamic parameter.

As mentioned previously, the baselines are plotted as functions of P_3/δ_1 and the changes in the other parameters are all computed as the percentage difference between the actual measured value and the baseline value, taken at the constant measured value of P_3/δ_1. The nature of thermodynamic corrected parameters is such that the baselines legitimately may be plotted as a function of any of the corrected parameters, and the measured changes calculated at the constant value of the chosen abscissa. An appreciation of the choice of P_3/δ_1 may be had by examination of the "Fault Coefficient", Table 6.2. By definition the fault coefficients are the magnitude of the changes that will take place in the measured parameters, at a constant value of another specified measured parameter, in response to a 1% change in the independent six parameters, taken one at a time. For example, in response to a single fault of a 1% fall off in compressor efficiency, T_3/θ_1 will increase 0.59%, P_3/δ_1 will increase 0.42%, fuel flow will increase 1.92%, etc., com-

6.2.2 Continued

pared to their baseline values at constant $N_1/\sqrt{\theta_1}$. Also at constant $N_1/\sqrt{\theta_1}$, in response to a single fault of a 1% fall off in gas generator turbine efficiency, T_3/θ_1 will increase 0.15%, P_3/δ_1 will increase 0.46%, fuel flow will increase 2.55%, etc.

The fault coefficients are shown for baselines plotted as functions either of $N_1/\sqrt{\theta_1}$, P_3/δ_1, $W_f/\delta_1\theta_1^y$ or $HP/\delta_1\sqrt{\theta_1}$. At first glance the tables appear to be a collection of random numbers of arbitrary sign. However, upon closer study several unique observations may be made about the constant P_3/δ_1 table:

1. If only $N_1/\sqrt{\theta_1}$ increases and all others remain constant, then compressor pumping capacity has decreased.
2. If there is any change in T_3/θ_1, there is at least a compressor efficiency problem, even in a multiple fault situation.
3. If only fuel flow increases and all others remain constant, then burner efficiency has decreased.
4. If fuel flow, horsepower and T_9/θ_1 all go down, there is a power turbine problem.
5. If there is a large relative increase in horsepower and fuel flow of about equal magnitude in each, there is a gas generator turbine problem.
6. If there is no change in T_3/θ_1 but changes in the others, then there is a hot end problem.
7. If the only changes are a decrease in horsepower and an increase in T_9/θ_1, then power turbine efficiency has decreased.

6.2.2 Continued

In the event of a single engine fault these uniquenesses can be used to fortify the diangostic answers provided by the Gas Path Analysis, which of course can handle either single or multiple faults.

Beyond this uniqueness another advantage of using P_3/δ_1 as the plotting abscissa is the relative gains of the various resulting parameter curves. For example, when plotting parameters as a function of $N_1/\sqrt{\theta_1}$, the resulting curves are all relatively steep; therefore any small error in reading $N_1/\sqrt{\theta_1}$ will be magnified several times in the calculated error or change in the parameter. These same parameters plotted as functions of P_3/δ_1 exhibit considerably reduced slopes (reduced by the magnitude of the slope of the P_3/δ_1 vs. $N_1/\sqrt{\theta_1}$ curve); therefore calculation errors are considerably smaller than the corresponding errors using $N_1/\sqrt{\theta}$. In view of the above reasons, it was felt advantageous to use P_3/δ_1 as the plotting abscissa.

Thus proper application of the Gas Path Analytical technique will permit isolation of the condition of the engine to a degree which was heretofore not previously available. This technique can also satisfactorily accommodate a wide range of "new" engine variations. For instance, consider that the airflow range for any engine may vary from 90 to 110% of a many-engine average. The diagnostic limit could then be established at -20% for the poorer engine and -40% for the best engine with the result that maintenance would be required when airflow deteriorates to 70% of the nominal average. The full life of very good engines is thus obtained without producing a dangerous situation in "poor" engines.

6.2.3 Gas Path Diagnostic Results

6.2.3.1 Test Cell Analysis

The data obtained during the Phase B test cell runs yielded encouraging results. Certain component signatures could be identified and the effort was useful in readying the hardware for flight tests and in refining the software application of the analytical techniques. However, these refinements have resulted in obsolescent data and the remaining discussions will concentrate on the subject of primary concern; i.e., the flight data and analytical results.

6.2.3.2 Phase D Flight Test Analysis

6.2.3.2.1 Signature Identification

This section will discuss those tests in which known degraded parts were implanted in four engines for signature identification purposes. A discussion of the verification tests including results is presented in Section 7. The degraded part tests included 10 total engine gas path tests on the above 4 engines. Five transmissions had discrepant parts installed. Additionally one maladjusted fuel control was tested. The 5 transmission tests utilized a nominal engine and the engine was not subjected to a detailed analysis. This same engine LE16522, was utilized in the fuel control tests.

The implanted fuel control malfunction attempted to alter steady state and transient performance. This was done by adjusting the fuel control pressure regulator and speed droop cam. The steady state errors did not significantly effect the engine operation because of the inherent closed loop control reset performance. Thus, the fuel control pressure regulator adjustment had little effect on speed in steady state because the fuel control

6.2.3.2.1 Continued

operates on a closed loop governing droop line. The power turbine speed droop cam change is also corrected by the control operation. The attempt to diagnose any transient fuel control malfunction was not included in the Test Bed Program. A significant increase in the digital processor memory requirements would have resulted if the fuel control schedules were duplicated in the AIDAPS digital processor.

The remaining 10 engine tests included one set in which the digital processor was not functioning properly, and one set with inadequate hardware and software definitions. Thus a total of eight tests were available for a detailed gas path analysis. These tests included both degraded gas path components such as nozzles and a turbine and non-gas path components such as bearings. The intent here is to demonstrate that discrimination can be obtained between the two types of malfunctions.

6.2.3.2.2 Baseline Analytical Approaches

The three analytic approaches which were used in the detailed analysis involved modifications of the baseline definition and the data set to be employed. The flight test data was analyzed using baseline data as a function of corrected gas producer speed ($N_1/\sqrt{\theta_1}$) and corrected compressor pressure ($PS3/\delta_1$). The test cell data taken prior to installation in the aircraft was also used to clarify the flight test results. A certain amount of random scatter was noted in the flight test data. This scatter was produced by both normal noise and by a non-ideal steady state test. In either case, a deadband was investigated as a means of minimizing the effect of this scatter. The deadband was employed such that only perturbations in excess of the band were used to define the malfunction. The results of the diagnostic

6.2.3.2.2 Continued

investigations are summarized in Figure 6-3. This table lists the inserted component (and engine number) and the results of each analysis. Each block lists the detected malfunction in order of severity.

6.2.3.2.3 Good and Bad Parts Discrimination

One test of the diagnostic system involves the ability to distinguish between good and bad parts. This test was successful in that the imbedded bad bearings were not detected as gas path malfunctions. (Bearing diagnostics are discussed in Section 6.4 on Vibration Analysis.) Two of the four turbine problems were also properly diagnosed as defective turbines. The other two turbine malfunctions were not detected by either the flight test data or by using the test cell data. This leads to the conclusion that the degree of degradation was not sufficient to register as a firm malfunction. Finally, the compressor flight test did not register as a degraded compressor. However, this test was conducted during an early period of hardware and software adjustments and may well have been detected if it had been repeated after complete system development.

6.2.3.2.4 Test Cell and Flight Data Comparison

A comparison of the flight data analysis and test cell data analysis is useful in appreciating the diagnostic results. This comparison indicates (Fig.6-3) that the conclusion are essentially the same, with the above noted compressor test exception, and that multiple deviations were detected in several instances when only single malfunctions were expected. The validity of the airborne diagnostics and data is thus confirmed by the test cell data. A detailed

6.2.3.2.4 Continued

tabulation of the actual parameter variations is presented in Figures 6-4 through 6-7 on a per engine basis. The five measured or explicit parameters are listed for a speed or pressure baseline along with the resulting calculated or implicit variables. One fact that is apparent from a study of these tabulations involves the degree of severity of the degraded parts. Very few deviations exceeded a ±5 percent band and most were within ±2 percent of the baseline for that engine. The "no problem" band for the implicit variables was ±2.5 percent which is a very tight band. This indicates that a conservative approach was followed in the selection of degraded parts, and that acceptable performance could be expected from the system as further deterioration occurs.

The actual differences between the test cell and flight data variations are attributed to several causes. First, completely different instrumentation and data collection techniques were utilized. Second, the baseline and bad part runs in the test cells were not generally taken in the same cell which introduces an unknown instrumentation error. This area of concern was minimized in the helicopter installation by using a consistent sensor set as much as possible. Finally, very few steady state points were available in the test cell data. The AIDAPS MEU averaged data throughout a long flight in order to perform its analysis.

6.2.3.3 Gas Path Trend Data Analysis

6.2.3.3.1 General Background Information

The change in trend data collection flights, as outlined in paragraph 5.3.5.2, resulted in data sets obtained at intervals of 10 to 20 aircraft

6.2.3.3.1 Continued

hours. Short term trends are not presented for this reason. Moderate term trends (in the 100 hour time span) yielded little information and the major results to be discussed include all data in the long term trend.

The actual trend results are presented in Figures 6-8 through 6-17. Figure 6-8 illustrates the number of data frames which were used from each flight. This information served in conjunction with the parameter value and time to define the trend lines.

6.2.3.3.2 Measured Parameter Trends

The five measured parameter trend plots are presented in Figures 6-9 through 6-13. The plots for fuel flow (Figure 6-9), horsepower (Figure 6-10), and exhaust gas temperature (Figure 6-11) illustrate reasonable results. The fuel flow which is required to obtain a given pressure condition should increase as engine wear occurs and the power which is produced would also be reduced. The rising exhaust gas temperature is indicative of poorer efficiency and is consistent with the horsepower and fuel flow trends.

The data on compressor discharge temperature and gas producer speed (Figure 6-12 and 6-13, respectively) illustrate barely discernible trends. The most probable conclusion is that no trend has developed in these two parameters at this time.

The estimated life of these five parameters has not been evaluated. The five measured parameters are symptoms of an event and it is more correct to analyze the event rather than the symptom.

6.2.3.3.3 Calculated Parameter Trends

The turbine inlet temperature (Figure 6-14) illustrates a trend which is similar to the exhaust gas temperature, i.e., that increasing temperatures are occurring. However, the shallower inlet temperature slope indicates that the hot exhaust gas temperatures and less efficient turbines. The trend limit for T_{T5} is approximately +5%. The trend line would then intersect this limit at 4240 hours. The conclusion from this long estimated life is that only an initial deterioration has been encountered.

The gas producer turbine is characterized by the nozzle area and efficiency plots of Figures 6-15 and 6-16, respectively. The nozzle area is increasing which indicates that nozzle erosion is occurring and the turbine is becoming less efficient. Preliminary trend limits of ±20% for turbine area and -15% for turbine efficiency result in a life estimate for the gas producer turbine of 2900 hours (2880 hours for nozzle area and 2920 hours for turbine efficiency). This conclusion also indicates that only an initial wear pattern has been established.

The power turbine area trend (Figure 6-17) exhibits a somewhat steeper slope than the other turbine parameters and thus results in a shorter estimated life. The aircraft time to reach a 20% deterioration at the indicated slope is 1670 hours. The explanation of this shorter estimated life involves the number of parameters which could be measured as discussed in Section 6.2.2. The proper analysis of the power turbine would require instrumenting the pressure between the turbines which was not practical for the AIDAPS Test Bed Program. The parameter plotted in Figure 6-17 thus represents both the

6.2.3.3.3 Continued

area and efficiency of the power turbine. The gas producer turbine area and efficiency deteriorations of 20% and 15%, respectively, sum to 35% which, if used as the allowable power turbine deterioration, yields an estimated life of 2830 hours. This life then tends to indicate that the power turbine and gas producer turbine are deteriorating at the same rate.

The plots of compressor temperature (Figure 6-12) and gas producer speed (Figure 6-13) indicate that no marked trend has developed in either parameter. These two parameters are the major factors in compressor efficiency and airflow and thus no firm trend was identifiable in these calculations.

6.2.3.3.4 Data Dispersion

The standard error of estimate for the least squares fitting algorithm was evaluated to obtain a measure of the data dispersion and resulted in the following estimated life bands.

Parameter	Limit	Average Life	Min. Life	Max. Life
ΔT_5	7%	4250 hrs.	3850 hrs.	4630 hrs.
ΔA_5	20%	2890 hrs.	2650 hrs.	3130 hrs.
$\Delta \eta T$	15%	2910 hrs.	2790 hrs.	3040 hrs.
ΔA_2	35%	2820 hrs.	2710 hrs.	2940 hrs.

6.2.3.3.4 (Continued)

The limits for the gas producer turbine area (ΔA_5) and efficiency ($\Delta \eta T$) of 20% and 15%, respectively, were tentatively assigned utilizing engineering judgement and the limited information obtained during the test program. The nozzle area variation (Δ An limit of 35%) was set larger because an additional measurement is required to differentiate between the power turbine area and efficiency (Reference Paragraph 6.2.2). The final limits in a production system will include a more extensive data base and approval of the engine manufacturer.

The intent of the above table is to illustrate the quality of the data by estimating the average life as outlined in Figure 5-44 and then displacing the average trend line to intersect the highest data point for minimum life and the lowest data point for maximum life. The estimate component life in hours then varies from $\pm$ 9% for ΔT_5 to $\pm$ 4% for Δ An and indicates that the data dispersion is acceptable.

6.3 Analysis of Mechanical Diagnostics

The derivation of the actual mechanical logic and information which was used in defining the diagnostic limits is presented in Section 5.3. In general, the degraded parts, exclusive of any gas path or vibrational characteristics, did no produce a decisive diagnostic message. The lack of a diagnostic output is attributable to several possible reasons:

1. Sensor malfunction;
2. Incorrect limits;
3. Parts which were still serviceable; or
4. Parts which reflect an effect rather than a cause of the malfunction.

6.3.1 Discussion of Results

Sensor malfunction was considered as a reason for the lack of a diagnostic but was eliminated from the candidate list by confirming the sensor calibration and by exchanging the sensors between the two helicopters. A simple re-definition of the diagnostic limits would not make detection of the degraded mechanical parts possible. The operating levels of the bad part parameters on AC 61011 were essentially the same as that of good parameters on the trend aircraft, AC 17223.

The serviceability of the implanted part must certainly be viewed as a candidate reason for the inconclusiveness of the mechanical diagnostics. A conservative approach to the selection of degraded parts was noted in both the gas path and vibration analysis effort. The cause and effect relationship should be examined closely. Consider a gear box during its normal operation when the oil level decreases. Additional heat would then be

6.3.1 Continued

generated due to inadequate lubrication and chips would form and excessive wear would probably result. However, the process is not generally reversible and putting that worn part back in a clean lubrication supply of the correct amount should not be expected to produce high temperatures or chips.

6.3.2 Mechanical Parameter Trend Data Analysis

The general comments regarding the gas path parameter trends (paragraph 6.2.3.3.1) are also valid for the mechanical parameters with respect to the time span which was analyzed and the number of data frames in each sample (Figure 6-8). The actual mechanical parameter trend results are presented in Figures 6-18 through 6-32.

6.3.2.1 Electrical System Parameters (Figure 6-18 through 6-21)

The four voltages which were monitored during the AIDAPS Test Bed Program are:

1. the starting battery voltage, Figure 6-18;
2. the AC essential bus, Figure 6-19;
3. the AC instrument bus, Figure 6-20; and
4. the DC essential bus, Figure 6-21.

These parameters exhibited modest trends but fairly large dispersions around the trend line, therefore, significant life estimates are difficult to make. This dispersion could have been caused by load (current) variations, and these variations should really be considered if accurate trending is to be done. The estimated lifes are presented below

6.3.2.1 Continued

Parameter	Life	Uncertainty of Estimate
Start Battery Volts	1400 hrs.	±30 hrs.
AC Essential Bus	635 hrs.	±105 hrs.
AC Instrument Bus	710 hrs.	±160 hrs.
DC Essential Bus	875 hrs.	±100 hrs.

6.3.2.3 Hydraulic System Parameters (Figure 6-22 through 6-24)

The three hydraulic system parameters which were monitored for trending are:

1. hydraulic pump leakage flow;
2. hydraulic pump temperature rise; and
3. hydraulic supply pressure.

The health of the pump is primarily established by the leakage flow and temperature rise and these parameters exhibited an estimated life of 670 and 620 hours, respectively. However, the uncertainty of the data, as reflected by the tolerance band on the life estimate, is ±65 hours for leakage flow and ±175 hours for the temperature rise. Discussions with Bell Helicopter representatives also indicated that firm data as to what constitutes a worn pump is not presently available. A large uncertain area thus exists as to the differences between a new pump and a failed pump. However, the flow and temperature rise life estimates are in reasonable agreement and illustrate that these parameters should be monitored to indicate the status of the pump.

The hydraulic supply pressure data (Figure 6-24) exhibits a large variation and a low life estimate of 340 hours. However, the major data

6.3.2.3 Continued

sets before 160 hours are in the range of 650 to 750 psi and those after 160 hours occur in the range from 1000 to 1100 psi. This indicates that a control setting adjustment may be desirable. The preliminary conclusion then is that no marked trend has developed.

6.3.2.4 Fuel Pressure (Figure 6-25)

The fuel pressure data exhibits an estimated life of 950 hours. However, this life is strongly influenced by the data set at 230 hours which is near the end of the testing. The elimination of this point results in the dotted trend line and a life estimate of 4820 hours. The longer life is a more reasonable estimate but more flight data would have to be obtained and be required to confirm the validity of the assumption.

6.3.2.5 Transmission Lubrication System (Figure 6-26 and 6-27)

The transmission oil cooler flow (Figure 6-26) indicates a nominal trend towards increasing flow demand. The estimated life to the diagnostic limit is 800 ± 260 hours. This parameter and engine oil cooler flow should be monitored in a pre-production system to clarify this initial conclusion.

Transmission oil pressure (Figure 6-27) has been uniformly constant for the Test Bed Program. No marked trend has been established.

Transmission oil temperature has been significantly below the diagnostic limit of 230°F and was commonly below the calibration range of the MEU which was 100 to 300°F. No data is presented for this parameter.

6.3.2.6 Engine Lubrication System (Figures 6-28 through 6-30)

Engine oil pressure (Figure 6-28) illustrates a shallow trend with an estimated life of 1550 hours. However, this life estimate has been increasing as more flight data was obtained and the actual pressure has stabilized at 79.5 psig. Thus the most probable conclusion is that some initial wear was encountered but no marked trend has really developed.

The engine oil temperature exhibited the same characteristics as transmission oil temperature, i.e., the temperature was below both the diagnostic limit of 250°F and the MEU calibration range of 100 to 300°F. No data is presented on this parameter.

The bearing 2 temperature rise (Figure 6-29) exhibits an increasing temperature with time. However, a large portion of the data is above the diagnostic limit of 220°F and therefore no reasonable life prediction can be made. This does indicates that the bearing temperature differential should be monitored in a pre-production system to more firmly establish the diagnostic limit.

The temperature rise across bearings 3 and 4 (Figure 6-30) illustrates a trend towards higher temperatures. The estimated life to the diagnostic limit of 220°F is 440 hours, but this limit may require revision depending on the results of subsequent bearing tests as outlined above. It should also be noted that very high temperatures at 97 hours resulted from a hardware malfunction and were not included in the analysis.

6.3.2.7 Gearbox Temperature Differentials (Figure 6-31 and 6-32)

The temperature differential between oil and ambient temperatures was monitored for both the 42 degree and 90 degree gearboxes. This data is significantly below the 130°F diagnostic limit and no trend was established in the test bed exposure.

6.4 Test Cell Program

The following paragraphs elaborate upon test cell instrumentation, as well as the data analysis techniques and results.

6.4.1 Vibration Recording Equipment

The vibration data acquisition and recording system used on all phases of the AIDAPS program utilized an Ampex AR-200 magnetic tape recorder. In addition, the system contained provisions for signal conditioning, amplification, standardization of transducer signals, tape coding, and central control. A one-channel signal block diagram of the recording system is shown in Figure 6-33. The function of the various elements in the signal path will be outlined in the following paragraphs.

The diagram, Figure 6-33, shows a transducer connected to a device called a signal conditioner. Just as the name implies, a signal conditioner is used to condition the electrical output of a transducer prior to amplification and subsequent recording.

This conditioning may consist of balancing a DC-excited full bridge transducer, filtering the signal from a self-generating transducer such as an accelerometer or velocity pickup, or limiting or clipping the signal generated by a magnetic pickup used as a speed transducer.

In general, the signal conditioners are not used to modify the data such as integrating an accelerometer output to obtain velocity or converting an AC speed signal to record a DC analog of speed. The primary function of the signal conditioner is to condition electrical signals to make them most amenable to good recording techniques.

Signal conditioners also select a standardized source for electrical standardization of a recording channel. The term standardize is used instead

6.4.1 Continued

of calibration because fixed measured physical quantities are not inserted into the recording system as is done when calibrating a device. The recording system is standardized by inserting signals equal to zero and full scale recording level into the record system. On playback, the system is adjusted for zero output during zero standardize and full scale output reference voltage during full scale standardize.

The next block in Figure 6-33 is a pre-amplifier which is used to raise the signal amplitude from low level transducers to a level compatible with the input requirements of the record amplifier. Full scale output of the pre-amplifier is ± 2.5V pk.

A record amplifier is a device which converts and prepares electrical signals for optimum recording on tape. Both direct record and FM record techniques are used.

The record head converts the electrical signals from the record amplifiers into varying flux patterns on the tape.

6.4.1.1 Record System Implementation

Figure 6-34 is a photograph of the AR-200 recording system. To make the various groups of equipment a working system with central control functions and to make the system compatible with an automatic data reduction system, a number of control functions and coding signals are required. Figure 6-35 shows the major hardware elements of the tape recording system and indicates the routing of the system control functions. This is also shown, in more detail in Figure 6-40. Briefly, the Tape Junction Unit (TJU) houses the signal conditioners, the Tape Pre-amp Case (TPC) contains the signal pre-

6.4.1.1 Continued

amplifier, the Tape Control Unit (TCU) is the operational unit of the system which provides the system control and coding, the record electronics consists of the record amplifiers and associated power supplies, and the Time Generator provides time and run number coding for the tape.

6.4.1.1.1 Time Generator

A standard reel of instrumentation tape (10-1/2" reel of 1 mil tape) is 3600 feet long and represents a real time recording length of 25-100 minutes depending on tape speed. It would be a difficult task to locate areas of interest on the tape during playback if there were no coding, but just three quarters of a mile of tape filled with data. To resolve this problem on IRIG B Time Code Generator was used to provide indexing of the tape. This generator records a code on the tape in a standardized IRIG B format to define elapsed recording time in seconds and a run number. The run number is used to define groups of data points. The recording of this code makes it possible to automatically search and playback selected segments of data during data analysis.

6.4.1.1.2 Tape Control Unit

The Time Generator helps to solve one of the problems encountered in tape recording. However, more information in the form of coding is desirable for indexing a tape to eliminate confusion and allow for automatic playback technique. The Tape Control Unit generates various tone codes which are recorded for playback control information on a separate tape track. Code signals recorded in addition to the time code are as follows:

6.4.1.1.2 Continued

Record Code (3000 Hz)	Indicates duration of data recording and provides a precise .01% reference frequency.
Full Scale Standardize Code (1500 Hz)	Indicates the recording system is in the full scale standardize mode.
Zero Standardize Code (750 Hz)	Indicates the recording system is in the zero standardize mode.
Mark Code (500 Hz)	Used to indicate occurrences of interest during data recording.
"A" Code (315 Hz)	Indicates recording is from transducers at location "A", if location switching is used during testing.
"B" Code (250 Hz)	Indicates recording from location "B".

The listed code signals are multiplexed on track 13 using a direct record amplifier.

The Tape Control Unit also provides control signals for operating the various units of the Tape Record System. These control signals are routed through the system and serve to place the system in the various operating modes. A microphone input is also provided on the Tape Control Unit and voice information is recorded on track 14. Figure 6-36 is a photograph of the Tape Control Unit which shows the front panel operational controls and indicators.

6.4.1.1.3 Power & Control Junction Unit

The Power and Control Junction Unit provides the system power supplies and the routing of AC and DC power throughout the system.

6.4.1.1.4 Tape Junction Unit

The Tape Junction Unit (TJU) represents the "front end" of the Tape Record System. This unit provides a panel for connecting transducers to the

6.4.1.1.4 Continued

recording system, provides connections for transducer excitation sources, houses the Signal Conditioners, provides standardization sources, and routes the signals to be recorded to the Pre-amp Case. The TJU contains 12 slots for the Signal Conditioner plug-ins.

As was indicated previously, there are two standarize modes used to define the sensitivity and zero point of the record and playback system. These are designated zero and full scale standardize. For the velocity and acceleration transducers used on the program the full scale standardize signal is an absolute AC standardize signal (sine wave) at 200 Hertz with an amplitude of 30 peak millivolts. When the system is switched to the full scale standardize mode, attenuators in the pre-amplifiers and signal conditioners are automatically changed so that the input 30 peak millivolt signal is normalized to 2.5 volts peak at the system output.

Self-generating signal conditioners were used with both the velocity and acceleration transducers on this program. These signal conditioners performed the following functions.

1. Buffered the signals from the transducers to minimize any electrical loading.
2. Provide additional signal attenuation for those channels where the range of amplifier attenuator is not sufficient.
3. Provided a regulated 20 VDC voltage for the Columbia 1111-1 and CEC 4-128 accelerometers used on this program. (These transducers are piezo-electrical accelerometers with a built in stage of electronics. The electronics is essentially

6.4.1.1.4 Continued

a high impedance buffer stage composed of a field-effect transistor and associated components. The advantage of these transducers is that the output signal is at a low impedance level which minimizes the effects of cable capacitance and electrical noise inherent in high impedance transducers.)

4. Figure 6-37 is a photograph of the Tape Junction Unit.

6.4.1.1.5 Tape Pre-amp Case (TPC)

The primary function of the TPC is to house the pre-amplifiers required for amplification of low level transducer signals from the TJU. The pre-amp case output signals are routed to the Tape Record Electronics for recording on tape. Additionally, these output signals are also available on a monitor connector which allows all the recorded channels to be monitored on an oscilloscope so that assessments of data quality can be made. The following specifications apply to the AC pre-amps used on this program.

1. Input impedance - 2 [illegible] ms minimum
2. Output impedance - 50 ohms maximum
3. Gain - 58 db (830) (Full Scale input is ± 3 mv pk at ATT x 1)
4. Gain adjustable by attenuator in steps 1, 2, 5, 10, 20, 50, 100, and infinity. (The attenuator is automatically set to 10 during the Standardize Mode.)
5. Frequency Response - $\pm 1\%$ from 7 Hz to 50 KHz.
6. Dynamic Range - 46 db over full bandwidth

6.4.1.1.5 Continued

7. Overload Indicator which lights when peak output level exceeds full scale by 30 percent.

8. Output level meter which shows average output signal level in percent of full scale. Figure 6-38 is a photograph of the take pre-amp case, and 6-39 is a photograph of the monitor oscilloscope.

6.4.1.1.6 Record Electronics

The final block in the signal path in the Record Amplifier which prepares the electrical signals from the pre-amplifiers for recording on tape. As indicated previously, two recording techniques are used, FM and Direct recording.

The FM Record Amplifier is a device, with DC response, which converts input voltage to output frequency. The amplifier operates at a nominal or center frequency for zero input, and the frequency is modulated ±40% for full scale input signals (±2.5 volts).

The Direct Record amplifier mixes data signals with a bias signal and feeds the resultant to a record head. Full scale input, like the FM record amplifier, is ±2.5 volts.

The FM Record process is far the more precise than the Direct Record. Amplitude accuracy of the FM process is at least in order of magnitude better, and the dynamic range is approximately 10 db greater. The Direct Recording does, however, provide a decade greater frequency response than FM at the same tape speed.

6.4.1.1.6 Continued

In general, FM recording is used where DC or low frequency response is required and when good amplitude stability and linearity are desireable. Direct Recording is used only when frequency information is required or where wide bandwidth is mandatory. All vibration signals recorded on the AIDAPS program used the FM record technique at a tape speed of 15 ips. This provided a recording bandwidth of 7 Hz - 5 KHz for the vibration signals.

6.4.2 Test Conditions & Parts Implanted

The three operating conditions used in the test cell are listed in Table 6.3 (i.e. 2 torque levels at 6400 rpm, and one torque level at 6600 rpm).

Engine			Transmission				42°&90° Gearbc	
Speed rpm	Torque in-lbs	Horsepower	Torque %	Tailrotor load	Generator Amps	Horsepower	Torque in-lbs	Horse-power
6400	10,860	1100	135	Low	200	1008	3640	92
	8,860	900	128	High	200	987	1170	30
6600	10,504	1100	74	Low	150	570	2210	58

There were two considerations used in choosing these test conditions. The first was to choose conditions which were part of the present ARADMAC "green run" procedures. The use of "green run" test conditions minimized the cell time involved and allowed vendor testing to be performed without increasing the time a component normally remained in the test cell. In addition, the use of established "green run" test conditions meant the rig operators would be setting up conditions with which they were quite familiar and the chances of an improperly set up condition would be lessened. It was felt that if different test conditions had been chosen there would have been a greater chance for variations in test conditions from run to run. The second

6.4.2 Continued

consideration was to choose high power conditions which were representative of conditions likely to be seen during Phase D flight testing. The high power conditions were chosen to insure that a good vibration signature would be obtained.

Figure 6-69 is a summary of the defective parts implanted in the engine during the test cell phase. Figures 6-70, 6-71 and 6-72 are similar summaries for the transmission, 42° and 90° gearbox test cell data. Each summary gives the part name, part number, and part serial number of each part tested, as well as how many of a type were tested. A description of each defective part is also given along with the serial number of the component in which the defective part was implanted.

6.4.3 Vibration Data Analysis - Test Cell

6.4.3.1 Role in AIDAPS

One of the goals of the AIDAPS program was to determine if vibration analysis could be used as an effective diagnostic and prognostic tool. A secondary goal was to determine whether an analytically effective approach would lend itself to practical flight hardware implementation. In this regard, the entire power train system of the UH-1 helicopter was monitored using vibration transducers mounted at the selected locations. The power train system included the engine, the main transmission, the 42° gearbox, and the 90° gearbox. Detection of faulty gears and bearings was emphasized in the test program through the installation of worn parts in the power train. The following paragraphs deal with the general nature of the malfunctions that occur in bearings and gears.

6.4.3.2 Nature of Expected Malfunctions

6.4.3.2.1 Bearings

Any one of a great number of problems in machine design, faulty operation or maintenance, or improper environment can lead to bearing troubles. Some of the most common are listed below:

1. Lubrication Failure
2. Fatigue
3. Dirt
4. Passage of an electric current through bearing
5. Brinelling
6. Corrosion

However, even if a bearing is properly lubricated, properly aligned, kept free of dirt, moisture and corrosive agents, and properly loaded the bearing will ultimately fail from fatigue. Repeated stress cycles in heavily loaded contacts between rolling elements and raceways first result in microscopic cracks at the weakest points in the grain structure of the material. As these cracks propagate, small surface areas become loosened from the main body of the bearing material. Once such a fatigue crack or pit is formed it becomes the center for high stress concentrations and the initial fatigued area is rapidly increased. Ultimately complete failure of the bearing results. The other causes of premature failure listed above also have a similar effect on the bearing surfaces i.e. causing pits, scratches, race eccentricities, or other physical damage to the bearing. This damage causes the bearing to become noisy and to emit vibrations. The frequencies of these vibrations can be

6.4.3.2.1 Continued

calculated in a simple case. The calculations are based on the known bearing geometry and the speed of the shaft it supports. These frequencies, their definition, and derivation are shown below:

The following bearing parameters must be known to calculate the bearing frequencies:

d_1 = Outer Diameter of the inner race

d_2 = Inner Diameter of the outer race

d_3 = Diameter of the bearing element (ball or roller)

N = Number of elements in the bearing

$$D_1 = \frac{d_1}{d_1 + d_2}$$

$$D_2 = \frac{d_2}{d_1 + d_2}$$

The bearing frequencies are calculated from the above using the following relations:

F_r = Differential rotational frequency between the inner and outer race (shaft rps)

F_s = Element spin frequency = $F_r D_1 (\frac{d_2}{d_3})$

F_p = Element train passage frequency = $f_r D_1$

F_b = Frequency of a rough spot on the element = 2fs

Fi = Frequency of a rough spot on the inner race = $F_r N D_2$

F_o = Frequency of a rough spot on the outer race = $F_r N D_1$

6.4.3.2.1 Continued

The energy generated at these repetition rates is in general not sinusoidal but composed of waveforms rich in harmonic content. This phenomenon tends to spread the energy associated with a bearing malfunction throughout the frequency spectrum rather than confining this energy into discrete narrow bandwidths. The following discussion will illustrate this point.

Take the 42° gearbox as an example. The gearbox has no speed change between input and output shafts. It has a duplex input and output ball bearing plus a single input and output roller bearing. Further assume that the nature of the bearing malfunctions is unknown and that it could be either on the input or output bearing set. Table 6.4 lists the possible repetition rates/frequencies associated with a malfunction in this bearing assembly. If the malfunction is a fatigue pit on the outer race of the ball bearing (F_O) the vibration transducer would sense a waveform sililar to that shown in Figure 6-41

For purposes of this discussion assume the input and output shafts are

6.4.3.2.1 Continued

rotating at 4200 rpm. The fundamental frequency of F_o would be 346 Hz. However, due to the complex shape of this waveform, harmonic amplitudes exist at 2 F_o, 3 F_o, 4 F_o, etc. Also shown in Figure 6-41 is an arbitrary waveform representing F_r (70 Hz). This waveform is also complex and rich in harmonics. Harmonics in this waveform can be caused by shaft eccentricities, shaft unbalance, misalignments, etc. For purposes of this discussion it is assumed that significant harmonic amplitudes exist up to only 5 F_r. There is a complex modulation phenomenon that occurs between the basic shaft rotation frequencies (F_r, 2 F_r, . . . 5 F_r) and the F_o, 2 F_o, . . . 5 F_o frequency components. Figure 6-41 shows a superposition of these waveforms. This superposition or complex modulation causes a sideband structure to occur centered at F_o and multiples of F_o. This sideband structure orients itself as listed below:

$F_o \pm F_r$, $F_o \pm 2 F_r$, . . . $F_o \pm 5 F_r$

$2 F_o \pm F_r$, $2 F_o \pm 2 F_r$, . . . $2 F_o \pm 5 F_r$

$3 F_o \pm F_r$, $3 F_o \pm 2 F_r$, . . . $3 F_o \pm 5 F_r$

$4 F_o \pm F_r$, $4 F_o \pm 2 F_r$, . . . $4 F_o \pm 5 F_r$

$5 F_o \pm F_r$, $5 F_o \pm 2 F_r$, . . . $5 F_o \pm 5 F_r$

Similar sideband structures at different frequencies can be developed for F_i and F_b and are illustrated in Table 6.4. The table lists only the basic repetition frequencies and their sideband structure. It should be recognized that the above development was for a simple case of only a single pit on either the inner race, outer race, or bearing element. As the number of pits increases the modulation process becomes difficult to describe mathematically. The vibration energy associated with a large number of pits would tend to

6.4.3.2.1 Continued

distribute itself in the frequency spectrum in a complex manner. The emitted vibrations would tend to increase or decrease depending on the severity of the pitting. The exact level was to be verified during the test cell program for good and bad bearings.

6.4.3.2.2 Gears

Like bearings, gears also fail from a number of causes. Listed below are some of the more prevalent causes of gear failure.

1. Misalignment
2. Inadequate Lubrication
3. General wear
4. Plastic flow
5. Surface fatigue
6. Tooth breakage

The above malfunction sources cause much the same damage to gear teeth as was the case with bearings. Minute fatigue cracks form in the gear teeth, small particles of metal flake off the gear teeth, or the gear teeth are deformed under load causing the gearmesh to run rough and emit vibrations of increased amplitude. The level of gear vibration tends to be much higher than that of bearings due to the fact that load transmission is involved. Like bearings, a complex modulation process occurs for gears. Figures 6-42 and 6-43 indicate the fundamental gearmesh repetition rate (F_{GM}) at 4200 rpm for the 42° and 90° gearbox as well as the transmission. For the 42° gearbox, the fundamental repetition rate is 1936 Hz. Energy of this repetition rate is also complex and therefore rich in harmonics. Shaft eccentricities at F_r

6.4.3.2.2 Continued

and its integer multiples can cause the driving gearteeth to be driven into and away from the driven gearteeth resulting in a load fluctuation. The amplitude and phasing of tooth contact noise (1936 Hz) is increased and decreased and a complex modulation process occurs. Sideband structure are formed at the following frequencies:

$F_{GM} \pm F_r$, $F_{GM} \pm 2\ F_r$, . . . $F_{GM} \pm 5\ F_r$

$2\ F_{GM} \pm F_r$, $2\ F_{GM} \pm 2\ F_r$, . . . $2\ F_{GM} \pm 5\ F_r$

$3\ F_{GM} \pm F_r$, $3\ F_{GM} \pm 2\ F_r$, . . . $3\ F_{GM} \pm 5\ F_r$

The sideband structure is carried out only to $3\ F_{GM}$ to illustrate the point. Vibration data on the 42^o gearbox taken during the AIDAPS program shows that the sideband structure exists at least out to $F_{GM} \pm 10\ F_r$ and $2\ F_{GM} \pm 10\ F_r$. An analysis similar to the one above can be carried out for every other gearmesh in the UH-1 power train system.

6.4.3.3 Tradeoffs in Analysis Techniques & Bandwidths

All arbitrary broad-band signals can be considered to exist in three dimensions: amplitude, frequency, and time. A sinewave for example is an amplitude-time relationship that exists at some frequency. A plot of this sine wave time history on an amplitude versus frequency basis would result in a vertical line at the frequency of the sinusoid. Such a plot of the frequency content of an amplitude time waveform is called a spectrum or frequency analysis.

The characteristics describing this waveform could also be considered as a function of amplitude vs. time. One immediate result would be the original amplitude time function. If, rather than viewing this waveform in real time, another time function equal to the time difference between arbitrary sampled

6.4.3.3 Continued

points on the waveform were introduced and the characteristics of the original waveform were displayed as a function of amplitude vs. time delay or difference, a correlation function would be obtained.

On the other hand, if the arbitrary amplitude time signal were analyzed with respect to the percentage of time the signal exists within certain amplitude limits, a probability function would be obtained. Each of the above three functions: frequency spectrum, time correlation, and amplitude probability are important analytic tools used in describing the characteristics of complex broadband waveforms. However, the most widely used and generally understood presentation of signal characteristics has been a plot of its spectral content. Among the many factors responsible for the pre-eminence of spectral analysis are, (1) the successful history in utilizing Fourier analysis techniques in waveform analysis, (2) the availability of wave analyzers, and (3) more recently the availability of computer programs that allow waveform analysis to be accomplished using digital techniques.

Before deciding upon a specific technique and system to perform the vibration analysis on this program the following additional factors were considered. These factors outline the general characteristics the analysis system must possess.

1. The sheer volume of data (approximately 400 separate spectra) to be analyzed on this program dictated the analysis should be done digitally.

2. The general nature of malfunction information emitted by faulty gears and bearings, i.e., their anticipated frequency distribution indicated that a system with good frequency resolution was needed.

6.4.3.3 Continued

3. The system must be capable of analyzing an arbitrary broadband signal with both random and periodic components present.
4. The system should have a large dynamic range (60 - 80 db) to handle both the high level gearmesh vibration signals and the low level bearing signals.
5. Since the test program would generate data requiring many comparisons of good and bad parts in both test cell and flight operations an analysis system that would facilitate these comparisons was essential.
6. The system should be equally useful for diagnostics and prognostics.
7. The technique employed must be able to make comparisons between good and bad parts on both an amplitude and frequency basis. This is mandatory if fault isolation to a particular line replaceable unit (LRU) is to be accomplished.
8. The system should provide for increases in signal to noise enhancement capabilities in the event that normal operating background noise on a particular component obscures the desired malfunction signal information.

These considerations clearly indicated that narrow band spectral analysis would be required at least as an intermediate step in the data analysis process. Figure 6-44 graphically illustrates the usefulness of a narrow-band constant bandwidth analysis compared to other more coarse types of frequency analysis. In this figure it is assumed that a segment of broadband analog vibration data is analyzed by filters of various bandwidths. The top curve shows the overall vibration level when the signal is analyzed with octave band filters. As can

6.4.3.3 Continued

be seen this analysis lacks sufficient resolution to give any significant information about the frequency content of the broadband waveform. The second curve is a narrower band (1/3 octave) analysis of the same broadband waveform. Here some frequency peaks show up at the lower frequencies but this analysis also loses resolution at the higher frequencies. The third curve in Figure 6-44 shows the results of yet a narrower analysis bandwidth ±4% or constant percentage analysis. This analysis defines more frequency peaks in the spectrum but also suffers from adequate resolution at the higher frequencies. Finally the bottom curve in Figure 6-44 indicates the spectral content of the broadband wave when it is analyzed by a narrowband 2 Hz constant bandwidth filter. The content of the broadband signal can now be seen to be a series of discrete frequency components arranged in a systematic order in the frequency spectrum.

The narrow band spectral analysis technique used on the AIDAPS program involved the use of Power Spectral Density (PSD) units [G^2/Hz or $(in./sec)^2$/Hz]. This narrow band analysis was implemented digitally using a computer and a Fast Fourier Transform (FFT) algorithm.

Three steps are performed to extract the PSD content from any broadband complex data signal. First, the signal to be analyzed is introduced to a narrow band filter which is swept over the desired frequency range. Second, a power function (squaring operation) is formed from the narrow band filter output. In the third step, a normalized average product is formed by integration and division of this product by the effective filter bandwidth.

Mathematically the above operations are expressed:

6.4.3.3 Continued

$$P(f) = \lim_{\substack{T \to \infty \\ B \to 0}} \frac{1}{T} \int_0^T \frac{Ys^2}{B} \, dt$$

where: $P(f)$ = PSD

Ys^2 = instantaneous square of the signal within a narrow bandwidth

B = signal bandwidth

Closer inspection of the above equation reveals that in order to obtain a true PSD, data records and averaging time must be infinitely long and the bandwidth must be infinitesimally narrow. It is thus impossible to obtain a true PSD, rather in implementing practical PSD analyzers the PSD obtained is an estimate of the true PSD. The quality of this estimate is dependent upon compromises between data record length, averaging time, and analyzing filter bandwidth. An important point to remember is that the above considerations apply only for random data since the average power of a periodic wave is the same whether it is averaged over one or many periods. Since data collected on the AIDAPS program is a combination of harmonically related sinusoids due to various rotational speeds, sinusoids due to structural responses, and random signals, one of the first questions one might ask is how good must the estimate of the PSD for random data be and how is it best described.

The quality of a PSD estimate is related to the statistical degrees of freedom associated with a random data sample. The number of degrees of freedom (N) is defined as N = 2 BT, where B is the analysis filter bandwidth and T is the length of the broadband data sample that is analyzed.

It has been shown by various researchers that the output of a narrow band filter approaches a Gaussian distribution when broadband random noise is

6.4.3.3 Continued

applied at the input. This relation holds true irrespective of the distribution of the input signal. Since, as mentioned earlier for PSD work, the output of the filter is squared, the properties of the square of a Gaussian random variable are of immediate interest. The square of this variable follows a chi-square (X^2) distribution. The ability to specify or estimate the PSD is related to this chi-square distribution and the number of degrees of freedom associated with the narrow band analysis. Figure 6-45 shows curves commonly used in PSD analysis of random data that relate the quality of the spectral estimate to the number of degrees of freedom for various confidence limits. All narrow band analysis performed on the AIDAPS program utilized 64 degrees of freedom. If a 98% confidence level is assumed the curve in Figure 6-45 indicates that the true mean square value lies between 1.7 and .7 times the observed mean square value. In this case the observed mean square is the squared and averaged output of the narrow band analyzing filter for a particular increment of bandwidth within the analysis range.

More simply, imagine a series of similar data records from one transducer mounted on one component such as the 42° gearbox were taken and analyzed using a narrow band PSD technique. If the output mean square signal level of the same narrow band filter were measured for each of the data samples taken, and the data was truly random, that level could vary by a factor of $\frac{1.7}{.7}$ or $\frac{2.4}{1}$. (98% confidence and 64 degrees of freedom)
For a given statistical confidence level the quality of the PSD estimate improves as the statistical degrees of freedom increases. However, Figure 6-45 indicates that the rate of improvement diminishes rapidly as the degrees of freedom increase. The number 64 chosen for this program was a compromise betwee

6.4.3.3 Continued

required accuracy, required analysis bandwidth, and computer computation time.

The narrow band spectral analysis technique was implemented using a digital computer. Analog data was first digitized, and the digital representation of the analog signal was stored in the computer memory. The memory capacity, as well as the rate at which the analog data is sampled and digitized, are important considerations which affect the frequency range of the analysis and the frequency resolution, i.e., analyzing filter bandwidth.

With the Fast Fourier Transform (FFT) utilized, the nominal bandwidth (B) of the analyzing filter synthesized in the FFT algorithm is defined as the reciprocal of the length of the digitized data sample which is stored in the computer memory. The detailed shape of the filter synthesized is shown in Figure 6-46. The plot is normalized to the nominal bandwidth B. This filter has an asymptotic attenuation slope of 18 db/bandwidth octave.

Selection of the exact bandwidth for the analysis filter involves many considerations and some compromises.

1. Earlier the statistical degrees of freedom associated with a data sample was defined as

$$N = 2\ BT$$

If the analysis filter bandwidth B is made narrower and the degrees of freedom are to be kept constant then the length T of the digitized data sample must increase directly as filter bandwidth decreases. Other things being equal a reduction of analysis bandwidth by one half would double the computer memory requirements and the length

6.4.3.3 Continued

of time required to make a digital spectrum analysis. As a matter of economics, the analyzing filter bandwidth should not be narrower than required to properly resolve the significant components in a broadband data signal.

2. The selected analyzing filter bandwidth should be narrow enough to separate high level normal components in a broadband data signal from possible adjacent low level frequency components that are indicative of a malfunctioning gear or bearing. In order to make a satisfactory bandwidth selection some knowledge of the amplitude and frequency distribution of the data is required. The Bell Helicopter Company report, titled Automatic Inspection, Diagnostic, and Prognostic System (AIDAPS) Testbed Program - Task II was especially helpful in this area.

3. The analyzing filter bandwidth should be narrow enough to provide sufficient signal to noise enhancement so that low level bearing vibration signals can be separated from the normal operating background vibration level. Again this requires some prior knowledge of the expected nature of the broadband data in order to make an initial selection. The degree of deterioration of a particular bearing or gear which must be recognized may also play an important part in filter bandwidth selection. That is, as the part wears, the amplitude of the malfunction signals may increase relative to the amplitude of the background vibration level.

6.4.3.3 Continued

The following example serves to illustrate how a judicious choice of analysis filter bandwidth can provide significant signal to noise enhancement in the case of a signal obliterated by noise. Assume a sinusoidal signal of 10 millivolts rms exists within a random signal of 1 volt rms in a frequency band from 10 Hz to 2000 Hz. This sinusoidal signal could, for instance, be a bearing frequency component. If this composite signal were applied to an RMS meter, the meter would indicate approximately 1 volt rms and would be totally useless in an attempt to separate the signal from the noise. The RMS signal to noise ratio of this composite signal is -40 db. If this composite signal is applied to a 200 Hz filter which is centered at the frequency of the sinusoid the filter output would be the 10 mv rms signal plus an additional noise component. However, by reducing the spectrum of the noise signal from a band 1990 Hz wide to one 200 Hz wide the level of the noise has been reduced from 1 volt rms to 317 mv rms (1 volt rms $\approx \sqrt{(.317)^2 \times \frac{1990}{200}}$

The filter output signal to noise ratio is now approximately -30 db, a 10 db improvement.

If the filter bandwidth was reduced from 200 Hz to 20 Hz the filter output would still contain the 10 mv rms sinusoid but the noise component would be reduced to 100 mv rms. The output signal to noise ratio is now -20 db, a 10 db improvement. For every 20 db reduction in filter bandwidth the signal-to-noise ratio is enhanced by 10 db. Any desired degree of signal-to-noise enhancement is possible provided that a narrow enough filter bandwidth can be obtained.

6.4.3.3 Continued

In a similar vein, a high amplitude normal frequency component can obliterate signals in adjacent frequency bands. Reducing the analysis filter bandwidth can improve on the ability to discriminate between energy sources. Consider a normal sinusoidal component of 1 volt rms at 1000 Hz and an analysis filter bandwidth (B) of 20 Hz. If it is desired to detect a signal of 10 mv rms at 960 Hz, close examination of Figure 6-47 shows that it is not possible. When the filter is tuned or centered at 960 Hz the desired 10 mv signal would be swamped by a 100 mv component due to the 1000 Hz signal. If the filter bandwidth were reduced to 2 Hz but still remained centered at 960 Hz, the interfering 1000 Hz signal component would be reduced to approximately 1 mv since the effective attenuation of a 2 Hz filter would be approximately -60db at 960 Hz.

To graphically illustrate the above concepts of analysis filter bandwidth, degrees of freedom, and signal to noise enhancement and the additional spectral resolution possible using narrower filter bandwidths reference is made to Figures 6-48, 49, 50, and 51, actual test cell data obtained during this program.

These figures are analog PSD plots of an identical sample of data taken on the 42° gearbox during the test cell phase of the program. For this particular test condition, there were defective implants in both the 42° gearbox and the 90° gearbox. The 42° gearbox had a bad output ball bearing implant while the 90° gearbox had a defective gear implant. The transducer selected was a velocity pickup on the 42° gearbox (parameter #59). The following table lists the important differences between the four analyses.

6.4.3.3 Continued

Figure	Analysis Bandwidth	DEGREES OF FREEDOM
6-48	40 HERTZ	512
6-49	20 HERTZ	256
6-50	2.5 HERTZ	64
6-51	1.25 HERTZ	32

It can be seen from these figures that as the statistical degrees of freedom increase, the data scatter associated with the random vibration improves. This is most noticeable in those areas of the graphs between the major peaks. These areas represent the normal background level vibration which tends to be random in nature. The agreement between the maximum/minimum points of the scatter and the values predicted by the chi-square distribution chart referred to earlier (Figure 6-45) is excellent.

As the analyzing filter bandwidth is made sucessively smaller, the four graphs also indicate the improvement on spectral resolution. For example, significant readily distinguishable peaks in the 1.25 Hz bandwidth analysis are completely obliterated in the 40 Hz analysis.

To convert from PSD units (which in this case are in inches per second squared per Hertz) to RMS units of the transducer (inches per second RMS) the following relation is used.

$$\text{RMS inches/second} = \sqrt{\text{PSD x BW}}$$

Here PSD refers to the value of a particular peak on the graph and is read on the left hand vertical axis of the graph, and BW refers to the particular analysis bandwidth used in an analysis. Depending on which PSD analysis is being used (i.e. Figures 6-48, 49, 50, 51) the corresponding BW would be

6.4.3.3 Continued

either 40, 20, 2.5 or 1.25 Hertz. On the right of the charts are numbers that correspond to RMS transducer units in inches per second. These numbers were obtained by using the above relationship. Additionally where significant peaks occur, a change in attenuator setting was made. The full scale PSD value for these particular peaks should be adjusted accordingly, as indicated by full range values. For example, all figures show significant peaks at 1850 Hertz. This is the basic gear clash frequency for the 42^0 gearbox at this operating condition. Just above the peak the number .01 indicates that the full scale value for that particular band is (.01 inches/second) squared/hertz rather than .001 (in./sec) squared/hertz as is the case for the majority of the spectrum. This type of presentation is necessary due to dynamic range limitations in the analog equipment used to make the analysis.

The final compromise for bandwidth chosen for the majority of data collected is as listed below.

Frequency Range	Bandwidth B	No. of Spectrum Points
0 - 5 Hz Data (accelerometer outputs)	14.6 Hz	341
0 - 2 Hz Data (velocity pickup outputs)	5.9 Hz .	341

6.4.3.4 Statistical Approaches

The narrowband spectral analysis in terms of amplitude squared per Hz versus frequency is only an intermediate step in the final data analysis process. Not only must a practical method for making comparisons between good and bad vibration spectra be implemented but also the implementation of this method must take into account the following:

6.4.3.4 Continued

1. The manner in which the vibration spectrum for a particular monitored parameter differs between samples of known good components. For example, the expected level of dispersion between spectra for 10 known good engines would be a determining factor.

2. The change in vibration spectrum with either test cell operating condition or flight test operating condition.

3. The repeatability of spectra obtained from the same good component checked at different times following removal and re-installation in either the test cell or aircraft.

4. How best to arrive at a vibration spectrum that can be considered an "average" spectrum representative of all good components.

5. Effect of locations of transducers.

6. Correlation of the vibration spectral pattern with known specific mechanical events on the monitored components.

7. Establishment of vibration amplitude threshold that discriminates between normal and abnormal operation with a high degree of confidence.

8. Development of a data reduction and analysis technique that can make diagnostic decisions quickly and display any abnormalities in a convenient format, especially in view of the magnitude of data that is involved. (i.e., each of the spectral analysis performed in the program involved at

6.4.3.4 Continued

least 341 individual frequency/amplitude points or lines. Since over 4,000 individual narrow band spectra were processed during the program, 1,264,000 (4,000 x 341) separate pieces of data had to be manipulated and compared. The sheer volume of data made digital data analysis mandatory.

Since comparisons between vibration signals of known bad and good components were to be made, information would be required regarding the average properties of these signals, the probability of their occurrence, their amplitude distribution etc. These considerations clearly indicated an analysis based on the statistical properties of these signals would be required. The statistical approach established for this program is detailed in the following sections.

6.4.3.4.1 Generation of Means

Once the narrow band vibration spectra for a given sample size of known good components operating at a particualr test condition were obtained, a mean or average spectrum was calculated. This was accomplished by summing the individual PSD values of the same frequency bands in each individual spectra and dividing the sum obtained by the number of individual spectra used to generate the mean. As a result, the mean spectra also contains 341 frequency bands but the PSD amplitude in each band represents an average value. Figure 6-52 shows the mean calculation in detail.

6.4.3.4.2 Standard Deviation

Similarly the standard deviation (σ) is also calculated for each of the 341 averaged frequency bands in the mean spectra. This is shown in detail in Figure 6-52. The calculated standard deviation does not imply any knowledge of

6.4.3.4.2 Continued

the amplitude probability distribution of the signals at the output of a particular filter band and is independent of the exact distribution. However, assume the true amplitude distribution of the vibration signals of the various good components at a particular filter band output is known. In that case, definite confidence limits could be established for that band relating the probability that most of the sampled vibration amplitudes of known good components lie between specific amplitude limits. As an example, assume a normal or Gaussian amplitude distribution exists at the output of a particular filter. This assumption also tacitly assumes that the sample size is sufficiently valid to be able to approximate the true distribution for normal components. In this case setting an amplitude limit at 3 standard deviations above the mean value for that frequency band would insure that 99.74% of the vibration signal amplitudes in this frequency band would be within this limit. The 3σ limit could be used to indicate general wear or incipient failure of a suspect component.

6.4.3.4.3 Comparison Summary

The comparison summary is a tabulation of data used to show the dispersion of both known good and bad components from the mean spectra.

Initially, once a mean spectrum is obtained from a number of narrow band spectra for good parts, each of the good part spectra is compared band by frequency band against the mean spectra at selected multiples of the standard deviation. These comparison levels for each band are the mean value and the mean value plus 1, 2, 3, and 4 standard deviations. The results of this comparison are tabulated. The number of frequency bands exceeding the above threshold values are printed and a graphic display of their occurrence in the

6.4.3.4.3 Continued

frequency spectrum is generated. See Figures 6-53 and 6-54 for examples. This tabulation gives both a qualitative and quantitative measure of the dispersion between known good components on a frequency/amplitude basis. The interpretation of these printouts is covered in detail in paragraph 6.4.3.6. There is an important point to remember regarding these threshold values (say the 2 σ case). Even though an item spectra is checked in each frequency band at the 2 σ level the absolute value of this level can change significantly for each of the 341 frequency bands in the spectrum. For an accelerometer signal as an example, the actual "g" level for a 2 σ threshold level could vary from 0.1 to 30 g across the spectrum.

In a similar manner, each narrow band spectra of a known bad component is compared against the mean spectrum and a similar tabulation of comparison levels exceeded vs. frequency is made. The mean value plus 4 standard deviations (M + 4 σ) was initially established as an amplitude threshold above which a component would be flagged as abnormal. This choice was based on the statistical considerations outlined above and the comparison summaries of the known good spectra used to establish a mean spectra. This technique, since it tabulates both the exceedance of a predetermined threshold and the frequency band in which it occurs, also facilitates correlation of the exceedance pattern at the 4 σ level with the known vibration frequency geometry of the monitored component and isolation of the malfunction not only to a line replaceable unit (LRU) but possibly to a specific defect within the LRU.

6.4.3.5 Vibrational Analysis Implementation

The AIDAPS vibration data was recorded on analog tape in both the test cell and flight phases of the program and shipped to Hamilton Standard for analysis. The first step in the analysis process which is illustrated in Figure 6-55 was to verify that valid data was recorded on the tape. This was done by playing back the data and observing the outputs of the individual tape channels on an oscilloscope looking for any indications of invalid data such as intermittent data signals, spurious noise components or clipped waveforms.

The outputs of the individual tape channels were then filtered with a low-pass filter prior to being fed to the analog to digital converter. In the analog to digital converter the analog signals were sampled at a rate determined by the frequency range desired and the digital samples recorded on digital tape. A computer Fast Fourier Transform (FFT) Spectral Analysis Program was then used to perform a narrow band Power Spectral Density (PSD) analysis on the digitized data.

Following verification of the narrowband printouts from the FFT program, the data was divided into two groups "good" parts and defective parts. The narrowband analyses for the "good" parts were then segregated by transducer and test condition and a mean and standard deviation spectra generated for each. The mean of each analysis band is generated by summing the PSD values for each sample of that band and dividing by the number of samples. In a similar manner the standard deviation of each analysis band is generated by taking the sum of the squares of the difference between each sample and the mean, dividing by the number of samples minus one and then taking the square root. A computer printout was obtained for each mean and standard deviation spectra generated.

6.4.3.5 Continued

Following generation of the mean and standard deviation, the individual narrow band spectra from the defective parts were compared against the mean spectra using the computer. This involved comparing each individual frequency band for the defective part spectra against the corresponding frequency band of the mean spectra and determining, in discrete increments, how much higher than the mean spectra the defective part spectra was. A computer printout was obtained for all comparisons made. In addition, for the test cell data the comparison information was printed out in the form of an X-Y plot. The comparison summary printout indicated by how much and at what frequency the defective part spectra exceeded the spectra of the "good" mean. The comparison summaries were then analyzed to determine if a signature existed for each type of defective part.

In many cases a data sample will contain frequency components above the highest frequency of interest. If these frequency components are not attenuated prior to digitizing an error is introduced in the form of spurious frequency components in the analysis. It is worthwhile mentioning that care was taken in the digitization to prevent fold over or aliasing errors by use of presampling filters. The Nyquist sampling theorem states that the sampling frequency should be at least twice the highest signal frequency of interest. In practice a sampling rate of three times the highest signal frequency of interest is used. Sampling at a rate lower than this fails to consider the fact that the anti-aliasing filter does not provide an infinite rate of attenuation at the cut-off frequency, but rather rolls off at some finite rate. The anti-aliasing filter used in the AIDAPS analysis for instance,

6.4.3.5 Continued

rolls-off at a rate of 30 db/octave. The output filter in the analog magnetic tape reproduce system provides an additional 40 db/octave attenuation. The data is sampled at the rate of three times the highest signal frequency of interest for a period of time determined by the number of degrees of freedom, the analysis filter separation and the frequency analysis range.

The analysis ranges, anti-aliasing filters, sampling rates and data sample lengths used for the AIDAPS vibration data are shown in Figure 6-56.

The primary consideration which determined the analysis ranges, filter separations and noise bandwidths listed in Figure 6-56 have already been covered in detail.

6.4.4 Test Cell Data Documentation

Throughout the AIDAPS program care was taken to adequately document all testing and data analysis work. During this program over 12,000 vibration data records were recorded on 50 reels of analog tape and when analyzed by computer over five miles of computer printouts were generated.

Figure 6-57 is an example of a test log sheet for the analog tape recording system. Every data run taken in the test cell phase was indexed on this type of log sheet. This log sheet lists the various parameters needed for the data reduction work such as reel number, run number, run start time, tape system attenuators and transducer serial numbers. In addition, the test conditions and serial numbers of the components and defective parts under test are listed. Figure 6-58 is an example of a master data log sheet to which each test log sheet is referenced. This log sheet lists the tape speed; parameters being recorded and the corresponding tape track; transducer type, serial number, and sensitivity; the type of signal conditioning and preamplifier equipment used, and whether direct or FM recording was used.

Figure 6-59 is an example of a special AIDAPS vibration log sheet used for this program. This log sheet contains a description of the defective part implanted, the type and serial number of that part, and the serial number and total time on the component in which the defective part is installed. Additional information such as date, log sheet number and run number are provided to reference this log sheet to the analog recorder test log sheet.

6.4.4 Continued

As explained and discussed in the previous "Data Analysis" section, there were three different types of computer printouts obtained during the analysis phase; narrow band printouts, mean and standard deviation printouts, and comparison summary printouts. The following paragraphs explain the interpretation of the heading information on these various data printouts.

Referring to the heading of Figure 6-60 and 6-61 for a moment, "Item" refers to the component upon which the particular transducer being analyzed is mounted; engine, transmission, 42° or 90° gearbox, "S/N" refers to the serial number of the component under test, "Reel No." and "Run No." identify the specific analog reel and run being analyzed, and "Test Log Sheet No." refers to the analog tape test logsheet which documents this particular run.

"Parameter No." and "Parameter Title" specify which particular transducer on the component is being analyzed. "Spectrum No." is an identifying number given to each spectrum to facilitate searching through the digital tapes. These numbers run consecutively from 0001 to approximately 4000.

"Item Speed" refers to the input shaft speed of the item under test while "transmission input speed" lists the corresponding transmission input speed. Item speed for the engine, however, refers to the output shaft speed. "Torque" lists the torque of the item under test in percent or in-lbs. depending upon the particular convention used in the various ARADMAC test cells. "Frequency range" lists the analysis range and "analysis bandwidth" lists the effective squared bandwidth of the analysis filter.

6.4.4 Continued

The narrow band analysis is printed below the heading in ten columns. Column one labelled PNT is the filter or frequency band identification number. The second column labelled FREQ is a listing of the filter center frequency. Column three is a coarse plot of the amplitude of the g-level in each filter band, each asterisk representing a 3 db. change in amplitude. Columns four and five are the amplitude levels in peak g's and inches/sec. in each analysis band. Two systems of units were used because both velocity pick-ups and accelerometers were utilized and comparisons between transducers were more convenient with the units of one transducer converted to the units of the other. Columns six through ten are a continuation of columns one through five for the second half of the frequency range. Below the narrow band printout are listed the characteristic frequencies of the bearings and/or the gearmesh and sidebands of the gears closest to the transducer being analyzed.

Figure 6-62 is an example of a mean and standard deviation printout from the test cell. Referring to Figure 6-62, the first part of the heading lists the mean identification number and the narrow band spectra that composed the mean. This information facilitates searching through the narrow band printouts to find a specific spectra for a "good" component. The parameter numbers and their corresponding mean identification numbers for the test cell data are as follows:

6.4.4 Continued

PARAMETER NO.	TEST CELL PHASE
7	4201, 4202, 4203
4	4401, 4402, 4403
8	4601, 4602, 4603
45	0101, 0102, 0103
47	0301, 0302, 0303
49	0701, 0702, 0703
123	0501, 0502, 0503
125	0601, 0602, 0603
126	1101, 1102, 1103
129	0201, 0202, 0203
59	2101, 2102, 2103
61	2301, 2302, 2303
64	3501, 3502, 3503
66	3701, 3702, 3703

The second part of the heading for the mean and standard deviation printout is for the most part an exact duplicate of the information printed on the narrow band spectra. Tape reel number, run number and spectrum number are not listed because the mean is made up of several different reels, runs and spectrums. The "No. of Spec. Avgd" figure indicates how many spectra for "good" parts went into the mean.

5.4.4 Continued

Unlike the narrow band printouts where the filter output was tabulated in both g's peak and ips peak regardless of the transducer type, the mean printout is in terms of the units of the particular transducer. The first column to the right of the plot gives the PSD value of the mean in terms of average units^2/hz, while the next column gives the mean in units peak. The units referred to are those listed in the heading information under "units" and correspond to the type of transducer used. The mean printout plot is in terms of the PSD, with the highest PSD value obtained in the mean plotted on the extreme right and the remaining values plotted in 2 db. increments below this, for a total range of 40 db.

The standard deviation analysis printout (reference Figure 6-63) gives the coefficient of variance and standard deviation of the mean for each filter bandwidth. The standard deviation is tabulated in the units of the transducer while the coefficient of variance is listed in percent. The plot that accompanies the standard deviation printout is a plot of the coefficient of variance with a 0 to 400% scale in 20% increments.

Figure 6-64 is an example of a test cell comparison summary printout for parameter #59 "input quill and output quill bearing", for the 42° gearbox. This particular comparison summarizes the results for the defective ball bearings implanted in the 42° gearbox for one of the three test conditions. The mean identification number indicates which mean these defective parts were compared against. The "tape", "run", "channel" and "spectrum number" identify the particular defective part runs. Referring to Figure 6-64, the first run compared was run #314, spectrum #1069. This defective part

6.4.4 Continued

had 107 bands out of 341 exceeding the level of the mean. At the mean plus one standard deviation level there were only 67 bands which showed exceedances. The mean plus two standard deviation level shows 49 exceedances, the mean plus three standard deviation level 39 exceedances and the mean plus four standard deviation level 34 exceedances. The next two lines are a list of the band numbers of the 34 individual frequency bands which exceeded the mean plus four standard deviation level. The remaining runs are summarized in a similar manner showing the exceedances at each of the comparison levels and the band numbers for those individual frequency bands which exceed the mean plus four standard deviation level. Each comparison summary contains all of the data samples for a particular defective part. The information summarized at the bottom of the comparison summary is a repeat of the information contained on the mean and standard deviation printout. A total of 348 of these comparison summary printouts were generated for all transducers and test conditions of the test cell phase.

6.4.5 Summary of Test Cell Analysis

Using the statistical methods, mean generation, and comparison summaries as outlined in previous sections of this report, analysis of the test cell data yielded the following results:

1. In every case of a bad implant on either the engine, transmission, or gearboxes 4σ exceedances were observed when a comparison was made of a vibration spectrum for a "bad" part against the mean spectrum for "good" parts. In addition, 4σ exceedances were observed not only on that transducer closest to the known implant, but also on those transducers relatively far removed from the known implant. For example, on the transmission which was instrumented with seven vibration sensors, a faulty gear implant on the upper sun gearmesh caused 4σ exceedances on all seven sensors. This indicated a substantial amount of transmissibility between the various components within the transmission. The same phenomenon was observed in the case of transducers on the engine and gearboxes.
2. For faulty gear implants, the comparison summaries indicated that the 4σ bands exceeded were primarily associated with the basic gear clash of the particular gear and its sidebands. Also evident are the harmonic frequencies of the basic gear clash frequency and the additional sideband structures associated with the harmonics. However, for known bad gear implants there is no single frequency band that can be used as a consistent indicator of a faulty gear. Depending on the particular gear, the dominant 4σ frequency band might be the 5th and 6th upper sidebands, or the 1st and 3rd lower sidebands, etc. The comparison summary plots graphically show this trend.

6.4.5 Continued

For example, take the case of three different faulty sun gears whose fundamental gearmesh frequency is 645 hertz when the input quill is running at 6600 rpm, (reference Figures 6-65, 6-66). The comparison summary displays the combined 4σ exceedances of all three gears vs. the frequency bands in which these exceedances occurred. In no case do all four implants share a common frequency band at the 4σ level (i.e. no 100% frequency bands). The same trend was observed for gear implants in the 42^0 and 90^0 gearboxes.

3. In the case of bearing implants a distinction between engine bearings and transmission and gearbox bearings must be made. This important point is fully expanded in the following paragraphs. Realization of the necessary distinction, however, leads to two significant conclusions.

   1) For degraded transmission and gearbox bearings (at least for the degradation level utilized in the test cell implants), bearing frequencies can be picked up but are low in energy, and require more sophistication to detect than the concurrent and resultant changes in the gearmesh frequency and its sidebands.

6.4.5 Continued

2) For degraded engine bearings the bearing frequencies should be considered of great interest since fault levels are much higher in magnitude than in the case of the transmission and gearboxes, and since these faults may not manifest themselves in engine rotation frequency components (i.e., shaft unbalance of the N_1 and N_2 systems).

The above conclusions are amplified in the succeeding paragraphs.

a. Transmission and Gearbox Bearings

In every case of a known bad bearing implant in either the transmission or gearboxes, the spectral analysis performed indicated no 4 σ exceedances in frequency bands that could be associated with pits on the bearing inner or outer races or the rolling element. The pattern that was observed was similar to that for a faulty gear. For example, for a bad bearing implant in the 42° gearbox, spectral analysis and comparison indicated 4 σ exceedances associated with the gear clash frequency of the 42° gearbox. In addition, shaft unbalance components were found associated with the rotational speed of the shaft which the faulty bearing supported. This pattern was again observed for all bearing implants in both the transmission and gearboxes. A faulty main mast bearing, for example, indicated 4 σ exceedances associated with the upper sun and planetary gearmesh. This gear system drives the shaft supported by the main mast bearing. Since this trend was consistently observed throughout the test cell data, and as a result of data analysis to be subsequently referred to, it was concluded

6.4.5.(3)(a) Continued

that there was a mechanical reaction between the faulty bearing, the shafting, and the gears, causing the gearmesh to run rougher and emit increased vibrations. This reaction is not due to any pitting on the bearing raceways or rolling element but is caused by a general wear in the bearing assembly with attendant increases in shaft eccentricities and misalignments.

To establish the magnitude of frequencies associated with pits on the bearing elements used on this program, a sample of test cell data from the 42° gearbox was analyzed with filters of various bandwidths. This is shown in Figures 6-48, 6-49, 6-50 and 6-51. This 42° gearbox had a faulty bearing while at the same time the 90° gearbox had a faulty gear. (The 42° and 90° gearboxes were run as pairs). The spectrum obtained by the digital analysis is roughly equivalent to the resolution midway between Figures 6-49 and 6-50. The PSD plots of Figures 6-50 and 6-51 were generated using filter bandwidths of 2.5 hertz and 1.25 hertz. Figures 6-48, 6-49 and 6-50 show the spectrum to 5 KHz while Figure 6-51 shows the spectrum limited to 2 KHz. It should be noted that these spectral plots were generated from the same sample of analog vibration data. Inspection of these four curves indicates that as the analyzing filter bandwidth is decreased, more peaks can be distinguished in the frequency spectrum. In the 2.5 Hz and 1.25 Hz analysis an almost bewildering array of spectral peaks can be observed. The frequencies of these various responses have been labeled and identified. To determine which frequencies are associated with a bearing and which are caused by gearmesh the various frequencies must be sorted out.

6.4.5 (3)(a) Continued

The majority of high amplitude responses are associated with shaft unbalance frequencies, gearmesh, and gearmesh sidebands. Although these plots are obtained from transducers located on the 42° gearbox, it can clearly be seen that the gearmesh and sideband structure associated with the 90° gearbox is present. This indication of transmissibility between gearboxes on the test rig hinted at the potential transmissibility that might be encountered on the aircraft when all components of the UH-1 power train system would be coupled together.

To assist in identifying the major peaks and their sources, detailed knowledge of the frequency geometry of both gearboxes is required. The following tabulation lists the major frequency peaks and the sources of these components.

42° GBX SHAFT UNBALANCE and HARMONICS	$F_{r42} + 2F_{r42} + \ldots + 10F_{r42}$
42° GBX FUNDAMENTAL GEARMESH FREQUENCY AND SIDEBANDS	$F_{GM42} \pm F_{r42} \pm \ldots \pm 10F_{r42}$
42° GBX 2 ND HARMONIC GEARMESH FREQUENCY AND SIDEBANDS	$2F_{GM42} \pm F_{r42} \pm \ldots \pm 10F_{r42}$
90° GBX INPUT SHAFT UNBALANCE	SAME AS 42° GBX
90° GBX OUTPUT SHAFT UNBALANCE	$F_{090} + 2F_{r090} + \ldots + 10F_{r090}$
90° GBX FUNDAMENTAL GEARMESH AND INPUT SHAFT SIDEBANDS	$F_{GM90} \pm 2F_{r42} \pm \ldots 10F_{r042}$
90° GBX FUNDAMENTAL GEARMESH	$F_{GM90} \pm F_{r090} \pm \ldots 10F_{r090}$

6.4.5.(3)(a) Continued

As an aid in sorting the various spectral peaks on the graphs, the following table was made. It lists the various sources of vibration and identifies them with a code. This code must be used in interpreting the various spectral peaks on the graphs of Figures 6-48 to 6-51.

Codes for spectrum analysis Figure 6-48 to 6-51

A	42° gearmesh	
$A_{+1}, A_{+2}, A_{+3}, \cdots A_{+n}$	42° gearmesh upper sidebands	
$A_{-1}, A_{-2}, A_{-3}, \cdots A_{-n}$	42° gearmesh lower sidebands	
B	2nd harmonic 42° gearmesh	
$B_{+1}, B_{+2}, B_{+3}, \cdots B_{+n}$	2nd harmonic upper sidebands	
$B_{-1}, B_{-2}, B_{-3}, \cdots B_{-n}$	2nd harmonic lower sidebands	
C	90° GM gearmesh	
$C_{+1}, C_{+2}, C_{+3}, \cdots C_{+n}$	90° GM upper sidebands	input shaft
$C_{-1}, C_{-2}, C_{-3}, \cdots C_{-n}$	90° GM lower sidebands	input shaft
$D_{+1}, D_{+2}, D_{+3}, \cdots D_{+n}$	90° GM upper sidebands	output shaft
$D_{-1}, D_{-2}, D_{-3}, \cdots D_{-n}$	90° lower sidebands	output shaft
E	2nd harmonic 90° gearmesh	
$E_{+1}, E_{+2}, E_{+3}, \cdots E_{+n}$	90° upper sidebands	input shaft
$E_{-1}, E_{-2}, E_{-3}, \cdots E_{-n}$	90° lower sidebands	input shaft
$F_{+1}, F_{+2}, F_{+3}, \cdots F_{+n}$	90° upper sidebands	output shaft
$F_{-1}, F_{-2}, F_{-3}, \cdots F_{-n}$	90° lower sidebands	output shaft
$G_1, G_2, G_3, \cdots G_n$	Shaft unbalance (72 Hz)	
$H_1, H_2, H_3, \cdots H_n$	Shaft unbalance (28 Hz)	

6.4.5 (3)(a) Continued

For the item speed for which these PSD plots were generated:

F_{r42} = 68 hertz

F_{GM42} = 1832 hertz

F_{GM90} = 1017 hertz

F_{r090} = 26 hertz

Substitution of these frequency values in the above relationships allows the major responses to be identified on the spectral plot.

Figure 6-42 lists all the predominant sideband frequencies associated with the gearmeshes in the 42° and 90° gearboxes and the transmission. These frequencies were tabulated based on 6600 rpm at the transmission input quill as a reference. Any change in the reference speed of 6600 rpm will change the location of the gearclash and sideband structure for each gearmesh in the power train. For example, the gearmesh for the 42° gearbox based on a transmission input shaft speed of 6600 rpm as a reference is 1936 hz. A 1% change in shaft speed would change the gearmesh frequency by 1% or 19.4 hertz and cause the sidebands associated with this gearmesh to shift by a similar amount.

A speed correction technique was employed in the digital data analysis to correct for rig speed variations to within ± 1 analysis filter bandwidth. This speed correction technique coupled with the averaging spectra and the statistical analysis techniques employed in the data analysis was initially considered adequate.

6.4.5 (3)(a) Continued

Once all the gearmesh and sideband and shaft unbalance components in the spectrum were identified, a search for frequencies that could be related to the pit frequencies for the 42° gearbox bearings was made. Table 6-4 lists the frequencies that are associated with a pit on the inner and outer races and rolling element for both the ball or roller bearing associated with the 42° gearbox input and output shafts. Since there is no speed change in the 42° gearbox, both the input and output bearing frequencies are the same. The "boxed-in" frequencies in the table indicate the various pit frequencies. If no modulation was present, these frequencies and their harmonics would be the only frequencies present in the spectrum. The frequencies in the table have been corrected to 6238 rpm.

Inspecting the 1.25 hertz spectrum analysis of Figure 6-51, smaller responses can be observed throughout the spectrum including that portion where the pit frequencies would be expected. One significant observation regarding these smaller peaks is the frequency separation. This separation varies between 6 hz, 10 hz, and 15 hz and repeats itself systematically throughout the spectrum. If these components are the bearing pit frequencies, an explanation is required for this unusual spectral distribution.

Assuming a complex modulation process exists for the bearings as is the case for gears, the remainder of the frequencies in Table 6-4 were generated. In implementing the frequency structure in the figure, it was assumed that the basic pit frequencies were modulated in some complex manner by frequencies associated with the speed of the shaft which

6.4.5 (3)(a) Continued

the bearings support - in this case 68 hertz. In a manner similar to that for gearmesh sidebands, 68 hertz and harmonic multiples of this frequency were added to and subtracted from the basic pit frequency repetition rate for the various bearing elements. Fitting this frequency structure to that indicated in the spectral plot of Figure 6-51 gave excellent correlation.

Though not shown in the table, harmonics of the basic pit repetition rate can be modulated by the shaft rps and its multiples to produce a similar frequency structure higher in the spectrum just as the spectral plot indicates. Since the amplitudes of these frequencies that were tentatively associated with bearing pits are so low, a number of tests to insure the validity of the spectral data were performed.

To eliminate the possibility that inherent system noise in the record/reproduce data gathering and analysis process was a significant factor, a spectral plot of the recording and playback system was made with the system in the zero standardize mode. In this operating mode the transducer is disconnected from the recording system, the input to the signal conditioners is terminated in a low resistance representative of the transducer output impedance, and a tape recording is made. Spectral analysis of this recording will indicate the magnitude of the total system noise exclusive of the transducer. Figure 6-67 is a PSD spectral plot of the zero standardize mode data. The peaks between 0-1,000 hz are power line components, 60 hz and harmonics of 60 hz.

6.4.5 (3)(a) Continued

The maximum component at 240 hz is approximately 1 x 10^{-5} inches/second squared per hertz (PSD units of the transducer). Converting this to RMS units of the transducer using the relationship RMS units = $\sqrt{\text{PSD x BW}}$, a level of .005 inches/second RMS is obtained. This is the amount of power line component at this frequency (240 hz) that is coupled into the system. In a similar manner the average level of the system noise throughout the spectrum is about 1.2 x 10^{-6} (inches/second) squared/hertz. Converting this to equivalent RMS transducer units, a value of .001 inches/second RMS is obtained. This number represents the average noise level of the record and playback system and is the lower limit of signal detection. That is, any signal lower than .001 inches/second would be effectively mixed in with the system noise and irretrievable. Comparing the spectral plot of the zero standard-ize, Figure 6-67 with the 2.5 hz data spectrum, Figure 6-50 it can be seen that the system noise level is sufficiently below the level of the data signals associated with bearing pit frequencies.

In a similar manner, it is possible for magnetic tape recording and reproducing systems to distort recorded data due to speed variations commonly termed as flutter. This concern is especially important when trying to determine whether low amplitude level signals such as those observed for bearing pits are valid.

High amplitude signals recorded on tape are effectively modulated by the flutter and spurious sidebands related to the flutter frequencies are created. The amount of distortion that is created depends on the level of the modulated signal and the amount of flutter.

6.4.5 (3)(u) Continued

For example, the spectral analysis plot of Figure 6-51 (1.25 hz bandwidth) has many high level gearmesh and sideband frequencies in the spectrum. It is possible, due to poor flutter characteristics in either the tape recorder or reproducer, to generate new frequencies not associated with the data. To assess the significance of this effect and as a check on the record/reproduce system flutter, a spectral analysis was performed on a full scale standardize tape record.

In the full scale standardize mode, the vibration transducer is disconnected from the system and a fixed 200 hertz sinusoidal signal of a precise amplitude is inserted in all recording channels to calibrate the system.

Figure 6-68 is a spectral plot of a full scale standardize tape record using an analysis filter bandwidth of 1.25 hertz. The most obvious and highest level component is the calibration frequency at 203 hertz at a PSD level of .142 (inches/second) squared/hertz. The second harmonic of this calibration signal is also evident at 406 hertz. The amplitude of the 2nd harmonic is only 2×10^{-6} (inches/second) squared/hertz.

The sideband structure indicated in the spectral plot is caused by flutter in the tape record/reproduce systems modulating the 203 hertz signal. This signal is recorded at a full scale level and is about 10 times higher than the highest peak on the data spectrum of Figure 6-51. Accordingly, the level of the flutter sidebands should be decreased from 2×10^{-6} PSD units 2×10^{-7} PSD units in relating the flutter effect to the data spectrum. This level of 2×10^{-7} PSD units

6.4.5 (3)(a) Continued

is substantially below the overall noise level of the system. Consequently, flutter effects on the data were discounted.

A final test of the spectral data concerned itself with the random background vibration evidenced on the spectral plot. This random data in the spectrum can be most easily seen as the fluctuations between the major peaks of the gearclash sidebands.

To insure that the low level peaks associated with the bearing pit frequencies are not caused by random vibration fluctuations, limits must be established for the expected variation of the random data. It will be recalled from the previous discussion on PSD analysis that the true PSD value of random data is never obtained. The quality of the estimate is related to the statistical degrees of freedom of the data record and the chi-square distribution. Horizontal lines representing the limits of minimum and maximum excursion of the random data that would be statistically expected for the analysis in Figure 6-51 are indicated on the spectral plot. For the degrees of freedom of this analysis, these limits indicate a band, with 98% confidence within which the amplitude fluctuations associated with the random vibration will fall. The spectral peaks associated with the bearing pit frequencies are substantially above these limits.

6.4.5 (3)(a) Continued

The amplitudes of these bearing pit frequencies average approximately .003 inches/second RMS. This can be determined by looking at the right hand vertical axis of Figure 6-31 where the corresponding transducer units for a particular PSD value are shown. This level would not be detected by the coarser bandwidth used on the digital analysis.

b. Engine Bearings

The pit frequencies associated with the engine bearings were easier to identify than those on the transmission or gearboxes. The test cell data on engine bearings shows that the pit frequencies fall very close to those predicted by calculation. The data also indicates that the signal levels associated with the engine bearing pit frequencies are at higher levels than the signals obtained for transmissions and gearboxes. First, the pit frequencies are higher in the frequency spectrum giving more "g"s for a given displacement. Second, the centrifugal loading on the rolling elements increases as the square of rpm. For example, assume that the 42° gearbox and engine bearing rolling elements are the same distance from their respective rotating axes. The approximate six to one speed difference between these components indicates that the engine bearing rolling elements are subjected to 36 times the centrifugal loading of the 42° gearbox bearing rolling elements.

6.4.6 Summary of Test Cell Data

Figure 6-69 is a complete summary of the test cell data for one test condition on the engine. Figures 6-70, 6-71, 6-72 are similar summaries for the transmission, 42° and 90° gearbox test cell data. Each summary gives the part name, part number, and part serial number of each part tested, as well as the number of each type tested. A description of each defective part is given along with the serial number of the component in which the defective part was implanted. The number of bands which exceeded the 4σ threshold level are listed along with the band numbers of each of these exceedances. Not all of the summary sheets contain the band number information. For example, in Figure 6-69, there are no band numbers listed for the engine test cell data. This is due to the fact that the band numbers were not printed out as such, but instead the frequency information was printed out in an X-Y plot format for this component. This is also true for the defective gear cases for parameter #61 on the 42° gearbox. These summary sheets are simply a synopsis of all of the AIDAPS vibration log sheets and comparison summaries generated during the program.

6.5 Flight Test Program

6.5.1 Vibration Recording Equipment

The same equipment as described in Section 6.4.1 was also used during the flight test phase of the program with some minor modification. The system as described previously is basically a 12-channel system, but since additional recording capacity was required for the flight phase, an additional Tape Junction Unit and Tape Pre-amp Case were added to expand the system capacity to 24 channels. Since the tape recorder itself has only 14 tracks available for recording, the signals from each Tape Pre-amp Case were routed through an automatic switching box which sequentially recorded each of the TPC outputs for a predetermined time interval once the record mode was initiated.

6.5.2 Test Conditions and Parts Implanted

The flight test operating conditions are listed in Table 6-5. These conditions were chosen as being representative of the higher power steady state conditions flown during a typical mission profile. They were conditions which could be readily set-up and held by the pilot for the 30 second recording time.

Figures 6-94, 95, 96, and 97 (Flight Test Summary of Results) contain summaries of all defective parts implanted in the engine, transmission, 42° gearbox, and 90° gearbox during the flight test phase. Each summary contains the part name, number, and serial number of each part tested as well as the number tested. A description of the defective part and the serial number of the component in which the defective part was implanted is also given.

	Engine		Transmission
Test Condition	torque-psi ranges	N1 speed - % ranges	speed-rpm ranges
ground	24 - 26	92 - 94	6550 - 6612
hover	22 - 31	87 - 95	6505 - 6622
115 kt level flight	30 - 39	92 - 95	6505 - 6574

Table 6-5 Flight Test Operating Conditions

6.5.3 Data Analysis - Flight Test

6.5.3.1 Result of Using Test Cell Means

The data that was recorded during the first few flights on ship 66-1011 was analyzed and compared against the cell mean to determine whether or not the test cell mean could be used during the flight phase for comparison summary purposes. The comparisons of the good flight data against the test cell mean are illustrated in Figures 6-73, 74, 75, and 76.

When compared against the test cell means, the flight test data indicated a number of exceedances at the mean plus four standard deviation level. The 90° gearbox showed the greatest number of exceedances with an average of 194. The engine and 42° gearbox showed an average of 20 and 23 respectively and the transmission the least number of exceedances with an average of only 9.

These large number of exceedances at the mean plus four standard deviation level indicate that "good" parts installed in the aircraft experience higher vibration levels than the same parts when installed in a test cell. The following reasons are listed to explain this effect.

1. The mounting configuration is substantially different on the aircraft from the test cell. In the aircraft the parts are interconnected to each other and mounted to a rather flexible aircraft structure. In the aircraft there is intercoupling of vibration frequencies between components, the 42° and 90° gearboxes interact with each other and the transmission interacts with the 42° gearbox.
2. There is a general vibration level in the aircraft due to aero-dynamic effects and various rotating machinery such as pumps, generators, inverters, etc.

6.5.3.1 Continued

With the vibratory level of "good" components in the aircraft so much higher than the test cell components it was apparent that a new baseline would have to be generated for the aircraft.

6.5.3.2 Generation of Flight Test Mean

The initial baseline data gathered for the flight test consisted of six samples of each of the power train components. The individual transducer means generated from these six baseline samples did not exhibit the same statistical variation as the corresponding means generated during the test cell phase. A typical mean from the test cell exhibited coefficients of variance ranging from 50% to 250% whereas the corresponding flight test mean showed coefficients of variance ranging from 10% to 100%.

The low coefficients of variance for the flight test were indicative of very little "scatter" about the mean. This indicated that the flight test mean comprising six samples was statistically different than the samples that composed the test cell mean. For this reason six additional baseline runs were made following the verification testing. A new set of means containing the twelve baseline samples was then generated and the coefficients of variance for these means was found to be in the range of 60% to 300% with the exception of the transmission, reference paragraph 6.5.3.3. The larger coefficients of variance obtained with the larger baseline sample indicated the flight mean was now exhibiting statistical variations similar to the test cell means.

6.5.3.3 Initial Data Findings

Initial attempts to analyze and correlate known bad implants according to spectral patterns, using the statistical techniques outlined at the 4σ level produced puzzling results. Not only did 4σ exceedances occur on those transducers on the particular defective LRU but also on every other transducer in the power train system. It was initially concluded that two major factors were responsible for causing this effect.

The first factor was the transmissibility of vibration signals from one LRU to another. It was evident from the test cell data that a substantial amount of transmissibility existed between the two gearboxes in the test cell. In the flight test, since all the components were now intercoupled, it seemed reasonable to expect that increased vibration signals due to an implant in one LRU also could be sensed by transducers on the other LRU's. Some idea as to the expected degree of transmissibility that exists, or some means to nullify its effect would be required.

The second factor that could account for so many 4σ exceedances on all the transducers is related to the sample size of data used to generate the flight test mean and standard deviation spectra. Initial comparisons between known bad and good parts were made using a mean spectra for the transmission in which only two different LRU's were used. In order to expand the sample size quickly, several runs at similar operating points of a particular LRU were obtained and these several runs averaged together to form a mean. Statistically this procedure is poor. The standard deviation and coefficient of variance spectra for the above mean spectra bore this out. Coefficients of variance (ratio of the standard deviation to the mean value for a particular spectral line or filter) for these mean spectra ranged from about .05 to .35. Statistically this indicated a very tight dispersion between the various narrow band

6.5.3.3 Continued

spectra used to establish the mean. Comparing the coefficients of variance for the flight test mean with those established in the test cell also indicated this effect.

There is an inference to be drawn from this limited sample size. In order to test the flight implants statistically at the same confidence level as in the test cell, the threshold value of 4σ must be increased to reflect the limited sample size.

Accordingly, next, the analysis technique was tried with the threshold level adjusted to indicate exceedances at integer multiples of the standard deviation up to 10. Another problem arose in making comparisons between good and bad parts at the various threshold values up to 10σ. The nature of this problem is best defined by using a specific example. When a known bad 42° gearbox was implanted on a particular flight, the statistical comparison technique indicated 4σ exceedances on all of the monitored vibration transducers on the aircraft. Increasing the threshold value in steps of 1σ up to 10σ did cause the number of exceedances on all transducers except those on the 42° gearbox to drop to zero. At the same time the number of exceedances (say at the 10σ level) on the 42° gearbox transducers was reduced from perhaps 25 to two or three. For five known bad gearboxes, these two or three exceedances at the 10σ level were not in the same portion of the frequency spectrum. In other words, 10 to 15 different frequencies at the 10σ level were all indicative of a gearbox malfunction depending upon the particular gearbox of the five involved. The same general pattern was observed for known bad implants in the other power train components. Two or three frequency bands out of a possible 341 which changed position in the frequency spectrum for malfunctions of a given type on the same component was questionable for use as a highly reliable criteria to flag a part as bad.

6.5.5 Continued

At this time, it was evident that additional vibration spectra on good components would be required to increase the sample size of the mean spectra. These additional samples were not obtained until after the verification phase of the flight test was concluded.

When these additional samples of known good components were obtained, a new mean and standard deviation spectrum were generated. Comparisons were then made between the new mean spectra and those from the unknown bad implants in the verification test and the known bad implants. These new attempts at correlating the 4σ exceedance pattern to a known bad implant with a high degree of probability were not encouraging; the same basic problem as described above existed but to a lesser degree. Some cases of known bad implants could be readily distinguished while others could not be distinguished. The fact that multiple faults were implanted in the gear boxes further complicated matters.

The mean generated for the transmission transducers posed some special considerations based on the statistics of the analysis. Adequate sample sizes were obtained during the flight test for the engines, the 42° gearbox, and the 90° gearbox by changing these components as complete LRU's. This was not done for the transmissions because of the time required to replace transmissions. The mean for the transmissions was based on interchanging sufficient known good quantities of input quills, tail rotor quills, and main mast bearings in the two aircraft transmissions provided for the flight test. Inspection of the mean and standard deviation spectra for the various transducers on the flight transmissions indicated that the dispersion of the various good narrow band spectra used in calculating these means was still too tight. That is, the coefficients of variance for the various bands in the spectrum were too low (.05 to .50 in the majority of cases). This indicated that in order to

6.5.3.3 Continued

establish a satisfactory mean spectrum for the various transmission transducers, vibration spectra on a series of complete good IRU's would be required. The fact that the transmission means were too tightly dispersed manifested itself by producing significant numbers of 4 σ exceedances (75-200) on the transmission transducers even though there were no bad implants within the transmission.

It was then decided that study should be made into the vibration phenomenom and the problems associated with transmissibility in order to better interpret the data. These studies will be now covered.

6.5.3.4 Re-examination of Statistics

The calculation of the standard deviation is a straightforward arithmetic calculation. Further this calculation is independent of any statistical distribution. Additionally, it will be recalled that an index of dispersion, called the coefficient of variance (C.V.) is simply the ratio of the standard deviation to the mean value for a particular filter band output. Since each mean spectrum covers a 0-5 kHz spectrum and was for our analysis divided into 341 separate filter bands, each mean spectrum involves the calculation of 341 means, 341 standard deviations, and 341 coefficients of variance. Re-examination of the tabulated C. V. for each of the mean spectra for both the flight test and test cell indicated a range from about .1 to 3, indicating a substantial amount of dispersion within certain frequency bands. For example, a frequency band with a C. V. of 3 indicates a substantially wider range of amplitude values for that same frequency band in the various narrow band vibration spectra used to generate the mean compared to a filter band with a C.V. of .1 or even 1.0.

Without knowledge of the true amplitude distribution of vibration amplitudes at a particular frequency band output all that can be said regarding the above observed C.V.'s is that some bands in the mean spectrum have a wider distribution of amplitude values than others.

If the true amplitude distribution in the various filter bands was known, then the standard deviation and C.V. could be used to relate the amount of dispersion between the various filter bands of the mean spectrum. Since the exact nature of the amplitude distribution at the 341 filter band outputs is unknown and cannot be accurately determined from the data obtained due to the

6.5.3.4 Continued

small sample sizes, a normal or Gaussian distribution of amplitudes was assumed for all the 341 filter frequency bands. For example, take the case of the 42° gearbox frequency at 1933 hertz which is located in filter band 133 in the 0-5 kHz spectral analysis. If a number of known good gearboxes were spectrally analyzed, the assumption of a normal distribution at the output of filter band 133 indicates that the distribution of these vibration amplitudes associated with the 42° gearmesh would be Gaussian. The same reasoning can be extended to all other 340 filter bands in the spectrum.

Due to the small sample size involved (approximately 10) in calculating the mean spectra, the Cumulative-t-Distribution tables were used to compare the anticipated amplitude distributions with those actually obtained. The use of these "t" tables rather than the normal or Gaussian tables takes the small sample sizes into account in deriving confidence limits for the amplitude distribution.

As an example of the use of the "t" tables as opposed to the normal or Gaussian tables consider the following example based on a sample size of 10. Ten different gearbox vibration amplitudes from 10 different "good" gearboxes in the same frequency band are averaged and the standard deviation for that filter band obtained. The Gaussian tables indicate that 99% of all other good gearboxes compared against the mean generated using 10 samples would have vibration amplitudes in the particular frequency band under consideration that do not exceed the mean value plus 2.33 standard deviations. Using the "t" tables, however, indicates that for the same degree of confidence with a sample size of 10, the upper limit must be increased to the mean value plus 2.82 standard deviations. In a similar manner, as the sample size used to

6.5.3.4 Continued

calculate a mean and standard deviation decreases to two samples, the "t" tables indicated that the upper limit, for the same degree of confidence, must be increased to the mean plus 3.18 standard deviations. This example demonstrates the importance of an adequate sample size for the calculation of a mean and standard deviation spectrum.

There is a simpler approach to assess the dispersion of each "good" narrow band spectrum when it is statistically compared to its mean spectrum. That is to inspect and determine the number of filter bands out of 341 that exceed the mean spectra at the various testing levels and relate the amount of these exceedances to the assumed distribution. For example, for a sample size of ten and again referring to the Cumulative-t-Distribution tables, only 4% of the 341 bands or approximately 13 should exceed the mean plus 2σ threshold value. For convenience the assumed confidence level for the above example was taken at 96%. In a similar manner, at the same confidence level, only 3 bands out of 341 should exceed the mean plus 3σ threshold value. Re-examination of the mean data for both the test cell and flight test in light of the above discussion showed that for every calculated mean spectra, approximately one-half of the "good" narrow band spectra fell substantially outside of the expected distributions. One example of this from the test cell data for the 42° gearbox will be discussed in detail. Figures 6-77, 6-78, 6-79 and 6-80 are the tabulated and plotted comparison summaries for the 42° gearbox. Figure 6-77 lists the number of 42° gearboxes that were used to generate the mean spectrum at this operating condition. This figure also lists the number of exceedances each gearbox shows at various levels when it is individually compared to the calculated mean spectrum. If the number of frequency band exceedances over 13 at the 2σ level as outlined previously is used as a statistical dispersion criteria, 5 of the 11 gearboxes are non-Gaussian.

6.5.3.4 Continued

Another indicator of non-Gaussian distribution is the number of gearboxes that exceed the mean value in more than 170 frequency bands. Only one gearbox Serial No. B13-5199 exceeds the mean value in more than 170 frequency bands, whereas 5 or 6 of the gearboxes should. Further, gearbox no. B13-5199 exceedances at the various testing levels are so great compared to the other gearboxes that it is clearly not representative of an average good component.

Figure 6-78 is a graphic display of the same information tabulated in Figure 6-77. Here, all the components (11 gearboxes) are displayed simultaneously in terms of percent of all gearboxes versus frequency at the various testing levels. Each test level represents the comparison of 3,751 (341 x 11) individual data points. At the mean comparison level, for a normal distribution, one would expect 50% of the gearboxes to exceed the mean in at least one-half (170) of the total frequency bands in the spectrum. That this is not true is plainly evident by visual inspection. If straight vertical lines are drawn from frequency bands exceeding the 3σ level to the same frequency bands at the mean level, it can be seen that only one gearbox out of 11 exceeds the mean value of that frequency band. This, again, is strongly suggesting that one gearbox, with a high level vibration signal in one band distorts the mean value of that band to such an extent that the other 10 good gearboxes do not exceed the mean value of that band.

Figure 6-79 is a tabulation similar to that of Figure 6-77 except that the gearboxes tabulated in Figure 6-79 have known defective implants (gears in this case). The tabulation lists only the frequency bands that exceed 4σ, the initial threshold level established as a demarcation point between known good parts and known defective parts. Figure 6-80 is similar to Figure 6-78 in that the known bad gearbox spectral band exceedances when compared to their corres-

6.5.3.4 Continued

ponding mean spectral bands are simultaneously displayed in terms of per cent of all bad gearboxes (6 in this case) versus frequency at the test levels indicated.

The 4σ bands in the tabulation of Figure 6-79 for each known bad implant in an individual gearbox can now be correlated against the known frequency geometry of the 42° gearbox. At the indicated item speed the fundamental gearmesh frequency should occur at 1878 hertz with a sideband structure above and below this frequency with each sideband separated by approximately 69 hertz (the shaft rotation speed in RPS).

The "good" gearboxes whose frequency spectra are not representative of a normal distribution cause distortion of the mean spectra. This results in significant energy changes in certain bands associated with a malfunction to be overlooked in the statistical comparisons at the 4σ level. The significance of the frequency bands that show signal levels in excess of what is statistically expected is that they are related to some abnormality in the particular component. As an example, consider a known good 42° gearbox. This gearbox is determined good by inspection of the individual components gears, bearings, shafts, and gearcase by various techniques, i.e. visual inspection, dimensional checks, magnafluxing and Zyglo. However, once all the individual parts are assembled into a complete gearbox it is reasonable to expect that some gearboxes due to a stack-up of tolerances, shaft or bearing race eccentricities, or pinion eccentricities would run rougher than others and, consequently, emit vibration signals of increased amplitude. For the case of a gear, certain frequency bands associated with the gear clash rate, the side band structure, and shaft rotation would have an increased output amplitudes. If ten 42° gearbox spectra are to be averaged and even one or two of these boxes emits increased vibration sidebands in a portion of the frequency spectrum where the

6.5.3.4 Continued

other gearboxes have no sideband energy the net effect is a distortion of the statistics. The amount of distortion is directly related to the amplitude of the vibrations and the number of frequency components

The first effect of including an abnormal component in the calculation of an average and standard deviation spectra is an increase in both these quantities which is not representative of the average distribution. There is also an increase in the coefficient of variance (C.V. - defined earlier as the ratio of the standard deviation to the mean value for a particular frequency band in the mean spectrum in terms of PSD quantities). The C.V. is also tabulated for each frequency band of the mean spectrum. This quantity ranges from approximately .2 to 2.5 within each mean spectrum. Invariably the data shows the C.V. tends to be highest in those frequency bands that are associated with a malfunction of a particular part i.e., side band and gearclash frequencies and shaft unbalance frequencies for a gear. To quantitatively relate the effect of this distortion on the 4σ threshold level a relationship is required that allows the absolute level of the 4σ threshold to be calculated preferably in units of the transducer. This relationship is:

$$T.L. = M\left[1 + N(C.V.)\right]^{1/2}$$

where T.L. = Threshold value at any given standard deviation

M = Mean value of a filter band output in transducer units

N = Standard deviations (1, 2, 3, 4 etc.)

C.V. = Coefficient of variance

Below is a table derived using the above relation for various C.V.'s and standard deviation levels

N Std Deviation Level	TL(CV=2.5)	TL(CV=2.0)	TL(CV=1)	TL(CV=.5)	TL(CV=.2)
4	3.31 X M	3 X M	2.24 X M	1.73 X M	1.34 X M
3	2.91 X M	2.65 X M	2 X M	1.58 X M	1.26 X M
2	2.45 X M	2.24 X M	1.73 X M	1.41 X M	1.18 X M
1	1.87 X M	1.73 X M	1.41 X M	1.22 X M	1.09 X M

6.5.3.4 Continued

This table vividly demonstrates in a quantitative manner the effects of statistical distortion on the computed mean spectra. For example, the 4 sigma threshold value in terms of transducer units could range from 1.35 mean to 3.3 mean value for a particular frequency band depending on the C.V. in that band. The above indicates that if a mean spectra filter band is distorted statistically so that the C.V. in that band is 2.5 when a 0.2 value is more representative of the vibration amplitude of the average population for that band, the 4σ threshold value in absolute terms in raised significantly. The statistical distortion would also raise the mean value for a filter band with a C.V. of 2.5 relative to that same band with a C.V. of 0.2 approximately by a factor of two. In this case the vibration amplitude associated with an abnormality instead of indicating a 4σ exceedance at 1.35 mean value for that filter band would have to rise to an amplitude level of 2 x 3.3 mean value or 5 times $= \left(\frac{2 \times 3.3}{1.35}\right)$ that level statistically expected to be required to flag a part as abnormal. The above example is admittedly a worst case analysis but it is used to demonstrate the nature of the problem. The same effect exists in other filter bands to lesser degrees and is dependent on the amount of statistical distortion in these bands.

The distortion discussed above has two major effects on the data. First, it raises the 4σ threshold level in absolute transducer units sufficiently in the very frequency bands that are the best indicators of a malfunction of a part. Consequently, even though vibration amplitude levels do rise significantly in these bands, they do not rise sufficiently to exceed the 4σ threshold level that was established. Secondly, since most of these significant frequency bands that are associated with an abnormality within a given component are not indicated in the comparison summary, correlation of a known bad part with its frequency spectrum or signature is made much more difficult.

6.5.3.5 Highly Resolved Spectrum Analysis & Transmissibility Evaluation

Test cell data obtained from the 42° gearbox was selected to be used for initial high resolution spectral analysis for the following reasons:

1. The 42° gearbox is a simple system to analyze in that only one gearmesh is present. Any significant results obtained from this gearbox analysis could be extended to a more complex system such as the transmission which has several gearmeshes.
2. Test cell data was chosen because the engines and transmissions were run in separate cells during this testing and transmissibility between these components and the gearboxes was not a factor. However since both 42° gearboxes and 90° gearboxes were simultaneously tested in the test cell rig some idea of the amount of transmissibility between these two components could be obtained.

Figures 6-50 and 6-51 are two highly resolved PSD plots of the 42° gearbox data. Figure 6-50 is a 2.5 hertz analysis and Figure 6-51 is a 1.25 hz analysis. The transducer from which the vibration spectra were obtained was #59, a velocity pickup. The 90° gearbox had a bad gear implant and the 42° gearbox a bad output ballbearing implant when this data was obtained. Inspection of these two PSD spectral plots shows the following major spectral responses:

1. The basic 42° gearmesh frequency at 1832 hertz with a sideband structure separated by the input shaft speed of 68 Hertz.
2. The second harmonic of the 42° gearmesh frequency at 3664 hertz with a sideband structure separated by the input shaft speed of 68 Hertz.

6.5.3.5 Continued

3. The 90° gearmesh frequency at 1017 hz with a sideband structure separated by 68 hz and 26 hertz.

4. The second harmonic of the 90° gearmesh frequency at 2034 Hertz separated by 68 hz and 26 hertz.

5. Shaft speed components at 68 hz and 26 hertz and harmonic multiples of the shaft speed.

Since the 42° gearbox has no speed change between input and output shafting the sidebands associated with this gearbox are separated by 68 hertz. The 90° gearbox has a speed change, the input shaft rotating at 68 rps in this case and the output shaft at 26 hertz. This speed change between input and output on the 90° gearbox accounts for the double side and structure of the 90° gearmesh. One structure (68 hz) is associated with the input shafting, bearing, and drive gear, the other sideband structure (26 hz) is associated with the output shafting, bearing, and driven gear.

Also evident from Figure 6-51 is the second harmonic of the gearmesh and sidebands frequencies from the 90° gearbox mingled in with the basic gearclash frequency and sidebands of the 42° gearbox.

The gearmesh frequencies for the gearboxes are calculated by multiplying the shaft speed in rps by the number of gear teeth. The sidebands are separated from the gearmesh frequency by the shaft speed and multiples of the shaft speed. Since the gears have an integer number of teeth, sidebands of gearmesh frequencies for gearboxes coupled by shafts rotating at the same speed will fall in the same frequency bands if the sideband structure is extended far enough. For example, since the output of the 42° gearbox and the input of the 90°

6.5.3.5 Continued

gearbox are coupled by the same shaft, upper sidebands of the 90° basic gearmesh and lower sidebands of the 42° gearmesh will occur at the same frequencies. Figure 6-42 shows this effect and it may also be observed on the two spectral plots of Figures 6-50 and 6-51.

This factor is a source of interference which depends upon the absolute magnitude of a 90° gearbox sideband component, the transmissibility between the 90° gearbox and 42° gearbox at that frequency, and the relative magnitude of a 42° gearbox sideband compared to the transmitted magnitude of the interfering 90° sideband component.

For example, Figure 6-42 shows that the sixth lower sideband of the 42° basic gearmesh and the sixth upper sideband of the 90° gearmesh occur at the same frequency. For the example cited this frequency is 1424 hertz. This interference will occur even though the input shaft speed changes. Changes in shaft speed would only change the frequency at which the sixth sidebands occur.

6.5.3.5 Continued

Assume the true value of the sixth lower sideband of the 42° gearbox to be 0.5 g rms. If at the same time the true value of the sixth upper sideband of the 90° gearbox was 10 g rms and the transmissibility between the gearboxes was 0.1 at this frequency, 1 g rms (10 x 0.1) of the 90° gearbox response would be sensed by the 42° gearbox transducer. The net effect in the spectral analysis of the 42° gearbox vibration data would be to increase the apparent response in the 1424 Hz band from 0.5 g rms to $\sqrt{(0.5)^2 + (1)^2} = 1.12$ g rms. Thus the effect of transmissibility is to more than double the spectral energy for this example. A fairly accurate assessment of the amount of transmissibility between the gearboxes was obtained using the following technique: The mean spectra for the 42° gearbox and the 90° gearbox are first compared. The highest response for the 90° gearbox occurs at its gearmesh frequency. Comparing the magnitude of the response at this frequency band on the 90° mean spectra with the same frequency band on the 42° gearbox mean spectra yields significant information regarding the amount of transmissibility at that frequency. In a similar manner the highest responses on the 42° gearbox mean spectrum were compared with the 90° gearbox mean spectrum. Preliminary information indicated the transmissibility between the gearboxes in the test cell ranged from about .1 to .2 . Admittedly this information was obtained at a few selected frequency points in the spectrum and does not account for any potential structural resonances that might magnify the transmissibility at some frequencies. The above technique was also extended to the flight test data with similar results.

The net conclusion is therefore that transmissibility is predominant, and its value ranges from .1 to .2 . The effect of this transmitted

6.5.3.5 Continued

energy is variable depending upon the relative signal strengths of the interacting components. Its effects must be considered, however, in any analysis slanted toward fault isolation to the LRU.

6.5.3.6 MODES OF MODULATION

Further inspection of Figure 6-51 indicates that the sideband structure around the various gearmesh frequencies extends quite far in the frequency spectrum ($\pm$ 10 sidebands). Additionally, the amplitudes of the sidebands do not drop off rapidly as the frequency separation of a sideband from the gearmesh frequency increases.

Various researchers have described the sideband generation process for gears and bearings as related to amplitude modulation. For the case of gear sideband generation the physical process creating the amplitude modulation within the gearbox has been related to shaft, bearing race, and gear eccentricities. It has previously been assumed that these eccentricities add and create a net shaft unbalance at the shaft rps and harmonic multiples. This shaft unbalance then can cause the driving gearteeth to be driven into and away from the driven gearteeth resulting in a load fluctuation. This load fluctuation can then cause amplitude modulation of the basic gearmesh frequency which forms a sideband structure. However, shaft unbalance in the majority of cases manifests itself in vibration components which are predominant at the shaft speed. The amplitude of the higher order components diminishes rapidly with frequency. Due to the nature of the amplitude modulation process the sideband structure should show this same trend. Figure 6-51 indicates that this is not the

Continued

tion, in fact just the opposite effect is indicated. Significantly, higher order sidebands have amplitudes greater than those of lower order. This trend and other inconsistencies in relating the observed spectral distribution from the test data to a predominantly amplitude modulation phenomenon led to the conclusion that the predominant modulation process involved in generating the gearmesh sidebands was due to phase or frequency modulation.

To establish the existence of FM modulation in the data, the following experiment was performed. A tape loop of a sample of the vibration data from the 42° gearbox was made. The signal from this tape loop was played back into an FM discriminator or demodulator. The demodulator input filter was centered at approximately 3800 hertz which is the frequency of the 2nd harmonic gearmesh of the 42° gearbox. This filter allows only that portion of the total frequency spectrum associated with sidebands of the 2nd harmonic gearmesh to be processed by the demodulator. The 2nd harmonic gearmesh and sidebands were chosen because the 2nd harmonic gearmesh vibrations were twice the amplitude of the first harmonic. In addition, the frequency spectrum around the 2nd harmonic gearmesh was relatively free of any interfering frequencies. The results of this experiment were that the shaft frequency and its harmonic multiples emerged at the output of the FM demodulator. This experiment confirmed that the 42° gearbox gearmesh frequency was being phase modulated at the shaft rps and its harmonic multiples.

Prior to proceeding, a brief discussion and explanation of the two modulation methods AM and FM will be presented.

A general alternating wave may be presented by the equation

$$(1) \quad f(t) = A \cos(wt + \phi) \qquad w = 2\pi f \; ; \quad t = \text{Time}$$

6.5.3.6 Continued

In amplitude modulation A is varied in accordance with the time variation of the modulating signal. In angle modulation the phase term ϕ is varied in accordance with the time variation of the modulating signal.

Amplitude Modulation

The general expression for amplitude modulation of a carrier with a single sinusoidal frequency is

(2) $f(t) = K (1 + m \cos w_m t) \cos w_c t$ where K is a system constant
w_m is the modulating frequency
w_c is the modulated frequency or carrier
m = modulation factor and denotes the fractional extent by which the modulation varies the carrier amplitude

Figure 6-81 shows a typical modulating signal and the carrier envelope variation corresponding to this modulation described by the above equation. Equation 2 can be expanded using trigonometric identities to the following:

$$f(t) = K \cos w_c t + \frac{M}{2} \cos (w_c + w_m) t + \frac{m}{2} \cos (w_c - w_m) t$$

The sinusoidally modulated carrier is shown to consist of the sum of three sinusoidal components of different frequencies, the original carrier plus an upper and lower sideband. This is also shown in Figure 6-81. The amplitude of each sideband frequency for a sinusoidally modulated wave is m/2, which has a maximum value of 1/2 for 100% modulation. Additional characteristics of the AM process are:

6.5.3.6 Continued

1) The number of sideband components for a single modulating signal as shown does not change if the amount of modulation (m) is increased. For the case shown, if the modulation was doubled, the amplitude of upper and lower sidebands about the carrier w_c would also double but the carrier amplitude would remain constant and no additional frequencies would be generated.

2) The amplitude of each sideband frequency for a sinusoidally modulated wave is m/2 which has a maximum value of 1/2 for 100% modulation. Relative amounts of power in the sidebands are:

$$P_{carrier} = 1$$
$$P_{lower\ sideband} = \frac{m^2}{4}$$
$$P_{upper\ sideband} = \frac{m^2}{4}$$

OR

$$P_{total\ sidebands} = \frac{m^2}{4} \times P_{carrier}$$

With 100% modulation the total power in the sidebands for sinusoidal modulation is one half of the carrier power. However for 10% modulation it is only .005 x the carrier power $(\frac{.1 \times .1}{2})$.

3) Increased energy in the sidebands is not obtained from the carrier, the carrier power remains constant regardless of the amount of modulation.

Frequency Modulation

Figure 6-82 illustrates a carrier frequency modulated by a repetitive sawtooth wave of period T. As the modulating wave increases in magnitude the carrier oscillates more rapidly or the frequency deviation increases. That is the frequency deviation of the carrier is proportional

6.5.3.6 Continued

to the amplitude of the modulating wave. The rate at which the carrier frequency changes is the rate of the modulating wave.

The general expression for a frequency modulated carrier is

(3) $f(t) = \cos (w_c t + \beta \sin w_m t)$

where w_c is the unmodulated carrier
w_m is the rate of modulation of the carrier
β is the modulation index and represents the maximum phase shift of the carrier during a carrier cycle.

β is further defined as

$$= \frac{\Delta f}{f_m} = \frac{\Delta w}{w_m}$$

and is defined as the ratio of the instantaneous frequency deviation of the carrier to the modulating frequency.

Equation 3 may be manipulated and expanded to the following form (the details of this expansion may be obtained from any text dealing with modulation theory).

(4) $f(t) = J_0(\beta) \cos w_c t - J_1 (\beta) \quad \cos (w_c - w_m)t - \cos (w_c + w_m)t$

$\qquad + J_2 (\beta) \quad \cos(w_c - 2w_m)t + \cos (w_c + 2w_m)t$

$\qquad - J_3 (\beta) \quad \cos(w_c - 3w_m)t - \cos (w_c + 3w_m)t$

where $J_n (\beta)$ denotes a Bessel function of the first kind which occurs in many physical problems. Tables are available to evaluate the magnitude of a particular $J_n (\beta)$.

6.5.3.6 Continued

Equation 4 indicates a time function consisting of a carrier Jo (β) cos wct and an infinite number of sidebands spaced at frequencies $\pm$fm, +2fm, etc. from the carrier. Note that this is in contrast to the AM case discussed previously where only the carrier and a single set of sidebands existed. (Again this assumes sinusoidal modulation for both the AM and FM cases.)

For the FM case the magnitudes of the carrier and sideband terms depend on β, the modulation index. This dependence is expressed by the appropriate Bessel function. This also contrasts with the AM case, where the carrier magnitude is fixed and not affected by the modulation and the amplitude of the two AM sidebands varies only with the modulation factor m. Or, in other words, the value of the carrier term in the FM case Jo (β) is always reduced when modulation occurs because Jo (β) is less than one for all values of other than zero.

Another important characteristic of the FM process is that the average power in the modulated wave is not changed by the modulation. The power in the sidebands is obtained by a reduction in the carrier power. Mathematically this is expressed

$$Jo^2 (\beta) + 2 \sum_{n=1}^{n=\infty} Jn^2 (\beta) = 1$$

for all values of β.

6.5.3.6 Continued

The entire science of gear design emphasizes the importance of achieving linear angular velocity in order to optimize the transmission of torque or power. Any one of a great number of factors that can cause dynamic variations in angular velocity will effectively cause phase or frequency modulation of the basic gearmesh vibration frequency.

The only factor contributing to variations in angular velocity of a gear that can be determined statically is the effect of tooth profile variations. Tooth profile position variation or "position variation" is defined as the deviation of a gear's tooth profile from true position. Dynamic or operating variation of the angle (θ) shown in Figure 6-83 will contribute to position variation inaccuracies. The degree of position variation is determined by the addition of all the factors affecting the tooth profile positions; some of which are:

1. Rotating variations
   a. Radial runout
   b. Lateral runout
2. Tooth-to-tooth variations
   a. Profile deviation from true involute
   b. Profile spacing
   c. Tooth thickness variations

Typical geometric gear variations are shown in Figure 6-83. It is important to appreciate that once a gear has been fabricated, it contains position variations due to manufacturing tolerances which are an inherent property of that particular gear. The effect of these variations on a

6.5.3.6 Continued

vibration signature does not become apparent until after the gear is mated with another gear. Also, when a gear is assembled into a complete system, additional position variations are encountered due to the tolerances of the installation. These variations are generally due to runout of the shaft and/or the bearing races.

When a gear system is transmitting power the dynamic loads on the gearteeth also cause position variation errors. Some of the dynamic factors are:

1. Elasticity of the gear material and the gear mounting.
2. Torsional deflection and beam bending of the gear shafting.
3. Gear tooth loading and resultand deflections.
4. Dynamic balance of the rotating components.
5. Temperature differentials within the gearbox.
6. The mass of the gear and shafting system.

All of the above factors then contribute to dynamic geartooth profile position errors and effectively phase of frequency modulate the gearmesh frequency or carrier.

The physical process which generates frequency modulation is demonstrated by the example in Figure 6-84. In this example, it is assumed that a gearbox similar to the 42° gearbox with 20 teeth per gear is under consideration. There is no speed change between input and output shafts and the shaft speed is 100 rps. The top half of Figure 6-84 shows the theoretically perfect case of no geat tooth profile position errors.

6.5.3.6 Continued

A vibration transducer monitoring this gearbox would sense only the vibrations caused by the meshing teeth transmitting power which occurs at 2,000 Hz (20 teeth x 100 rps). Only the basic gearmesh frequency is assumed to be of significance in this example. The figure shows one complete revolution of the shaft or gear. Accordingly the unmodulated wave at the top of Figure 6-84 shows 20 sinusoidal cycles or 20 pairs of geartooth mesh. Since the shaft rotates at 100 rps, the time for one complete shaft rotation is .01 seconds as illustrated. The time between successive positive peaks on the unmodulated sinusoidal gearmesh frequency is $.5 \times 10^{-3}$ seconds $(\frac{.01}{20})$ and is constant from cycle to cycle. Since one shaft rotation is equivalent to 360 mechanical degrees of the gear and there are 20 teeth on the gear, the mechanical angle between successive tooth profiles is 18° (360/20). In terms of the electrical output waveform of the transducer which senses gearmesh vibration, one electrical cycle (360° electrical) is generated for each fractional turn of the shaft equal to the mechanical degrees between gear teeth. One mechanical degree is equal to 20 electrical degrees (1/18 x 360).

The two bottom illustrations in Figure 6-84 show the case for a tooth profile position mechanical error of 1/6 degree peak mechanical. This error could arise from one of the many factors discussed above and is assumed to vary sinusoidally as the gear completes one revolution. The net effect of the position error is to change or modulate the instantaneous frequency of the 2000 Hz gearmesh frequency. This is also shows in Figure 6-82

6.5.3.6 Continued

The peak mechanical position error of 1/6 degree would cause a peak phase shift of a carrier cycle of 3.3 electrical degrees ($\frac{1/6}{X} = \frac{1}{20}$; x = 3.3° ELECTRICAL). Assuming an approximate 3.6° error rather than 3.3° for convenience, it can be seen that the net effect of the position error is to cause a peak frequency deviation of the carrir of 3.6/360 x 2,000 or 20 cycles. This frequency deviation is equivalent to the Δf term in the expression of the FM modulation indes $\beta = \frac{\Delta f}{f_m}$. Since the shaft speed is 100 rps, and the position error is assumed sinusoidal over one gear revolution, the effective modulating frequency fm is 100 rps. This gives β a value of 0.2. If the position error variation over one gear revolution is non-sinusoidal, the effective modulating frequencies will occur at 100 hertz and multiples of 100 hertz since the variation pattern is repeated at a 100 Hz rate. The exact distribution of the harmonics would depend on the particular position variation pattern within the gear system and the modulation index β. For β = .2, the only Bessel coefficients of significant amplitude are J_0 (.2) and J_1 (.2). Substituting the values of these coefficients into the expression for the FM wave gives:

$$f(t) = .99 \cos w_c t + 0.1 \cos (w_c - w_m) t + 0.1 \cos (w_c + w_m) t$$

$J_0 (.2) = .99$	$f_c = 2000$ Hz
$J_1 (.2) = .1$	$f_c + f_m = 2100$ Hz
	$f_c - f_m = 1900$ Hz

6.5.3.6 Continued

This is very similar to the expression for the case of amplitude modulation, i.e., a carrier term of approximately unchanged amplitude and two side-bands of equal amplitude spaced from the carrier frequency by the modulating frequency. In fact, if a spectrum analysis of this FM wave were obtained, it would be impossible to distinguish the resultant frequency structure from that obtained with amplitude modulation. However, as the FM modulation index increases, or higher harmonics of the basic gearmesh frequency are considered, some interesting effects manifest themselves.

Figures 6-85, 86, and 87 show graphically the effect on the FM sideband structure associated with the carrier or gearmesh frequency as the modulation index β is increased. These figures also illustrate the second harmonic component of the basic gearmesh frequency scaled to give an amplitude of twice that of the basic gearmesh frequency. The reason for selecting two harmonic components at this relative magnitude is that this pattern has been consistently observed for the 42° gearbox spectral analyses (i.e., the second harmonic component is twice the value of the first).

Figure 6-85 shows the case for $\beta = 0$; i.e., no phase modulation. At the top of Figure 6-85 is drawn a gear tooth pattern indicating perfect meshing conditions; i.e. tooth profile position errors do not exist. Below the gear tooth pattern is shown the resultant wave which includes the first and second harmonics of the gearmesh frequency. Below the resultant wave the individual gearmesh frequencies (first and second harmonic) are shown with no frequency modulation. And finally at the bottom of Figure 6-85 is shown the spectrum of the vibration signal for

6.5.3.6 Continued

the case of no modulation. Oly two frequencies at relative amplitudes of one and two units are present in the spectru.

Figure 6-86 shows the case when some FM exists due to a sinusoidal phase modulation or profile position error. Here it is assumed that $\beta = 0.5$ for the first harmonic. If, as in the previous example, the gearmesh frequency is assumed to be 2000 hertz and the gearshaft rps is 100, Δ fc of the gearmesh if 50 hertz ($\Delta f_c = \beta x f_m$). Also for $\beta = 0.5$ the significant Bessel coefficients are now:

J_0 (.5) = .94

J_1 (.5) = .24

J_2 (.5) = .03

Referring to the equation describing the FM wave and inserting the above Bessel coefficients, it can be seen that the sideband structure has expanded. The spectrum of the FM wave now contains the original carrier or gearmesh frequency at a reduced amplitude of .95 and two pairs of sidebands separated from the gearmesh at $\pm$100 hertz and $\pm$ 200 hertz at relative amplitudes of .24 and .03.

Similarly, Figure 6-87 shows the case when the modulation index associated with the first harmonic of the gearclash increase to one ($\beta = 1$). The significant Bessel coefficients are now:

J_0 (1) = 0.77

J_1 (1) = 0.44

J_2 (1) = 0.11

J_3 (1) = 0.02

6.5.3.6 Continued

As contrasted to the case of β = .5, this case of β = 1 indicates that the frequency spectrum contains the gearmesh frequency at 2000 Hz but reduced in amplitude to 0.77 and a sideband structure at ±100, ±200, ±300 hertz from the carrier at relative amplitudes of 0.44, 0.11, and 0.02. The spectrum is also shown in Figure 6-87.

The point of the above discussion is the fact that a single modulating frequency related to the shaft speed and a moderate amount of frequency deviation (Δf_c) which is related to the peak tooth profile position error can combine to cause an extended sideband structure around the basic gearmesh frequency. The greater the frequency deviation (Δf_c) or effectively, the greater the modulation index β the more extensive is the sideband structure. Further, and most important to the understanding of vibration specifics, it can be seen that as β increases the energy increase in the sidebands is accompanied by a corresponding decrease in the energy at the carrier or gearmesh frequency.

The effect of frequency modulation on the second harmonic of the gearclash is somewhat different than the effect on the fundamental. Recalling that one of the equations describing the FM wave was

$$f(t) = \cos (w_c t + \sin w_m t)$$

Where w_c = carrier or gearmesh frequency (first harmonic)

w_m = modulating or shaft rotation frequency.

The equivalent expression for the frequency modulated wave at the second harmonic of the gearmesh is:

$$f(t) \text{ (2nd harmonic)} = \cos (n\, w_c t + n \sin w_m t)$$

Where n = 2

$$fm = \frac{wm}{2\pi}$$

6.5.3.6 Continued

This equation indicates a time function at twice the carrier or gearmesh frequency modulated at a rate determined by the shaft rps (fm) with an effective modulation index n β which is twice that (n=2) for the case of the basic gearmesh frequency.

Since β, the modulation index, is defined as $\beta = \frac{\Delta f_c}{f_m}$ and fm is the shaft rps the effective modulation index (β) for the second harmonic of the gearmesh frequency is twice that of the first harmonic. This is shown in Figures 6-85, 86, and 87. The effect of this phenomenon is to cause a more extensive sideband structure at the second and higher harmonics of a gearmesh frequency. This is also shown in the spectrum plots in Figures 6-50 and 6-51. The appropriate Bessel coefficients for the second harmonic modulation idexes of β = 1 and β = 2 are listed below.

$\beta = 1$	$\beta = 2$
J_0 (1) = 0.77	J_0 (2) = 0.22
J_1 (1) = 0.44	J_1 (2) = 0.58
J_2 (1) = 0.11	J_2 (2) = 0.35
J_3 (1) = 0.02	J_3 (2) = 0.13
J_4 (1) = Not Significant	J_4 (2) = 0.03

The fact that the modulation index increases in direct proportion to the harmonic number of a particular gearmesh frequency implies the following:

Even though the modulation information associated with a malfunction contained in the carrier and sidebands of a gearmesh is the same at the fundamental gearclash as it is at the higher harmonics, the spectral distribution of this information will be different depending upon which harmonic sideband structure is analyzed.

6.5.3.6 Continued

The significance of the modulation index ($\frac{\Delta f_c}{f_m}$) is that the Δf_c term represents the net sum of the many factors previously outlined that contribute to gear tooth profile position variations on a dynamic basis. The identification of this FM effect and its characteristics also explains the reaction initially observed in the test cell phase between a bad bearing implant and the gear and shafting system. (i.e. the bearing implant manifesting itself by causing a change in the related gear sideband structure).

The above discussion of FM assumed, in all cases, a sinusoidal modulating wave. However, this assumption does not represent typical gear tooth profile position variations. Significant harmonics of the sinusoidal pattern do exist. The extension of the FM process with a complex modulating signal or profile variation pattern is considerably more difficult and cumbersome than the case for a single frequency. However, the general characteristics outlined above are still applicable. For a complex modulating signal the energy in the gearmesh sidebands would occur at known points in the frequency spectrum, i.e. at integer multiples of the gearshaft rps or fm. The sideband structure would not be symmetrical about the carrier. This fact has been established by various researchers on frequency modulation and has been consistently observed in the path.

The modulation index β is an extremely important indicator as to the amount of phase or frequency modulation that exists in a particular gear, bearing, or shaft system. This index would in general be different for a particular gear system (i.e. 42° gearbox, 90° gearbox, transmission tail rotor, transmission input quill, etc.).

6.5.3.6 Continued

It would depend on such factors as

1. The geometry of the gear (no. of teeth, size of gear, etc.)
2. The manufacturing tolerances for a particular gear (input quill, tail rotor, etc.).
3. The mass of the gear.
4. The rotational speed of the gearshaft
5. The mechanical load

Assume, for example, that gear tooth wear is the only factor under consideration. If a mean spectra representative of all new 42° gearboxes were obtained, the sideband structure for all samples that were used to generate the mean spectrum should be similar if the gears were cut by the same machine. As the gearteeth in these gearboxes wear as a function of usage, the relative tooth profile position error would increase, causing β to increase and causing the sideband structure to be altered in some predictable manner. However, due to the many factors in addition to wear influencing the tooth profile position error, it can be expected that the sideband pattern would differ depending on the cause of a malfunction. At this time it appears that it may be possible to correlate the sideband structure with a particular malfunction.

The FM effect also tends to substantiate the test results obtained based on the statistical analysis of the data. It will be recalled that statistical indicators employed to test the narrowband spectra used to compute a mean spectrum for a particular component indicated that some of the components were not representative of the mean and caused statistical distortion. Invariably the filter bands that were associated with gearmesh sidebands indicated the highest coefficients of variance.

6.5.3.6 Continued

This brings out two distinct possibilities: first, that those components that differed statistically from the mean were "bad" to the extent that they required replacement; or second, that these components represented different degrees of wear relative to the other components used to establish the mean. The data analyzed tends to substantiate the latter case. A further extension of this line of reasoning would indicate that it is distinctly possible, using the statistical approach outlined earlier, to segregate "good" components into various mean spectra. These mean spectra then would be representative of various degrees of wear. Perhaps comparison of a suspect component could be made with the various mean spectra to determine useful operating hours of life remaining. Additional testing, data analysis, and correlation would be required to determine if this is in fact possible.

There is one additional significant factor concerning the FM process. That is the equations describing the modulating process predict that as the modulation index β increases, the energy associated with the carrier or gearmesh vibration frequency decreases. In fact at some modulation indices the energy associated with the carrier will disappear completely and all the energy associated with the FM wave is contained in the sidebands. This effect is not normally noticeable until the modulation index exceeds approximately 1.5. To ascertain whether this effect could be observed in the data, Figure 6-88 showing expected modulation indices for the various gearmeshes in the power train was

6.5.3.6 Continued

calculated. The basis of the calculation was that the maximum Δf_c term in the modulation index $\left(\beta = \frac{\Delta f_c}{f_m}\right)$ was 1%. This number was considered representative of what could be expected for the various gearmeshes. The modulation index was calculated by taking one percent of the various gearmesh frequencies and dividing by the respective shaft speeds associated with a particular gearmesh. This was done for both the driving gear and the driven gear.

The figure indicates that based on the modulation index only the upper and lower planetary gearmeshes would be likely candidates on which to check the existence of this effect. Vibration spectra from all the other gearmeshes indicated that the predominant components are the gearmesh frequencies. This would be expected based on the value of their modulation indexes.

Figure 6-89 is a narrowband spectral (2.5 Hz analysis filter) analysis of the vibration signals from transducer No. 125; a piezo-electric accelerometer which was mounted on the transmission input quill. This spectrum is from flight test data, and the defective implant is an input quill ball bearing. The spectral analysis extends from 0 to 5,000 Hz. The highest amplitude response is the input quill gearmesh frequency which for the item speed in this analysis is at 3133 Hz. Sidebands associated with this gearmesh are identified on the analysis for both the driving gear and the driven gear. They are spaced at approximately ± 108 Hz and ± 51 Hz from the gearmesh. These sidebands are of a relatively low value compared to the gearmesh. This is to be expected for two reasons. The first is that the modulation index associated with this gearmesh is low ($\beta = 0.3$). The second is that,

6.5.3.6 Continued

since the defective implanted bearing had minor pits and scratches on the balls and races, it is not reasonable to expect these defects to react on the gear with any appreciable magnitude.

Also identified in the spectral analysis are the other transmitted gearmesh frequencies associated with the tail rotor drive and accessory drive. The tail rotor basic gearmesh occurs at approximately 1830 Hz and its second harmonic at 3660 Hz. Sidebands of these gearmeshes, though they exist, are not transmitted with sufficient energy to make them standout above the background noise at this transducer location. Therefore no attempt was made to identify these components. The accessory drive gearmesh at 2779 Hz is also identified but the same reasoning as above applies to its sidebands.

Further observation of this spectral analysis shows significant responses located throughout the spectrum. They have been identified and labeled. It is seen that these responses are associated with both the upper and lower planetary stage gearmesh frequencies, sidebands of the gearmesh, harmonics of these gearmesh frequencies and sidebands associated with the gearmesh harmonics. It is again pointed out that the energy associated with these responses is being sensed by an accelerometer on the input quill.

The upper planetary gearmesh at the item speed for this analysis is 628 Hz. This component is missing yet two sidebands spaced about 5 Hz apart at 633 Hz and 638 Hz are apparent. Based on the modulation index table, 5 Hz spacing of sidebands should be predominant. The responses at 592 Hz and 683 Hz are also sidebands associated with the 626 Hz gearmesh. The

6.5.3.6 Continued

second harmonic gearmesh frequency of the upper planetaries, is evident at about 1255 Hz. However, the amplitude of this gearmesh relative to its associated sideband structure is quite low. Also notice that the sideband structure is separated by intervals of about 5 Hz. The sideband structure at the second harmonic is more extensive than that at the first harmonic. This is to be expected since the effective β at the second harmonic is twice that of the first harmonic. The dominance of sidebands at 5 Hz is also to be expected because the modulation index for the 5 rps shaft is greater than 3 times that for the 17 rps shaft.

The next significant response occurs around 2000 Hz. The first harmonic gearmesh of the lower planetaries should occur at 1945 Hz. Again this response is missing. However a sideband structure separated at intervals of about 17 Hz exists around this frequency. Again based on the modulation index this is understandable.

The next significant response is associated with the 4th harmonic of the upper planetaries. This should occur at about 2448 Hz. Again no significant response exists at this frequency. However, notice the spacing of the sideband structure around 2448 Hz. It is in 17 Hz increments and multiples of 17 Hz. This is an effect not observed at the lower harmonics of this gearmesh. However, since the effective β for the 17 rps shaft on the upper planetaries is now approximately 1.5, sidebands associated with this shaft now proliferate as expected. Also notice where the 5 Hz sidebands associated with the upper planetaries are now located. They are at 2705, 2720, 2730, 2760 Hz. The effective β associated with the 5 rps shaft based on the assumption in Figure 6-86 is now greater than 5.

6.5.3.6 Continued

The same reasoning can be extended to the other responses in the spectrum out to 5KHz. In general, since the modulation index is increasing, sidebands move further and further away from the harmonic of the gearmesh frequency and there is a general intermingling of sidebands spaced away from the gearmesh frequency by 5 Hz and 17 Hz.

The fact that the high level responses associated with the two planetaries actually exist is shown in Figure 6-90. This figure lists all of the major spectral responses observed in the mean spectra that were generated for each transducer on the transmission and gearboxes during the flight test. The table lists the magnitude of each peak response for each transducer in peak units of the transducer and the filter band in which it occurred. The center frequency of each filter band in the spectrum is also listed. The transducers are referred to by their identification numbers. Since all spectral analyses on the transmission were carried out from 0 - 5,000 Hz with a 14.7 Hz analysis filter, it will be appreciated that the resolution of this filter is not sufficient to separate all of the 5 Hz sidebands.

Inspection of Figure 6-90 shows that all the major responses such as tail rotor quill gearmesh, tail rotor quill second harmonic, the input quill gearmesh, the accessory drive gearmesh, the upper and lower planetary gearmeshes and their sidebands are listed. The table also allows cross-correlation between the various transducers on the transmission so that transmissibility effects can be evaluated. The table shows not only that the transducers closest to a particular excitation source give maximum response but also that there is a significant amount of transmissibility of this excitation to the other transducers. The item speeds in this figure were digitally speed corrected to a reference of 6600 rpm input quill shaft speed within ± 1 filter band.

6.5.3.6 Continued

Figure 6-90 shows that both velocity pickups #47 (axial) and #123 (radial) mounted on the transmission top case are most sensitive to the planetary gearmesh vibration. It further indicates planetary sideband components sensed by these transducers at levels of .32, .42, and .57 peak inches/second in filter bands around 4,000 Hz (second harmonic lower planetary gearmesh). In terms of acceleration this amounts to 15 to 40 pk g's.

Study of the test data also indicates that the FM process exists in the case of bearing pit frequencies. It will be recalled that the highly resolved spectral analysis and discussion of the data on the 42° gearbox from the test cell phase established the existence of the bearing pit frequencies. Although these frequencies were at extremely low amplitude levels, the nature of the spectral distribution of these frequencies follows the same spreading pattern observed for the gears. This can be explained by the fact that the bearing elements, be they balls or rollers, are separated from each other by a cage retainer. Mechanical effects contribute to changes in the relative spacing of these elements (i.e., tolerances, wear, loading) will effectively cause time variations between impacts as the elements strike a pit in the bearing races. This will give rise to frequency or phase modulation just as was the case for gears. Two major distinguishing features for modulated bearing frequencies are present. First, the overall amplitude level associated with a pit frequency is much lower than is the case for gears; secondly, the frequencies associated with the bearing are not only a function of the bearing geometry and rotational speed but also of the contact angle and any element slippage.

In conclusion, it can be shown quite positively that the predominant modulation process that accounts for extensive sideband structure associated

6.5.3.6 Continued

with gears and bearings is frequency modulation. This allows a great deal to be learned about the health of a mechanical system by studying a detailed spectrum analysis. It also precludes the usage of simple "amplitude of fundamental" measurements as indicators of the health of a system.

6.5.4 Flight Test Data Documentation

Documentation similar to that for the test cell was also kept for the flight test phase of the program. Every data run taken was indexed on the analog tape system test logsheets, and these test logsheets were referenced to master data logsheets in the same manner as in the test cell. The special AIDAPS vibration logsheets were also kept listing the defective parts implanted, their serial numbers and the descriptions of the defects.

Analyses similar to those of the test cell were performed on the flight test data and the computer printout formats were similar to the test cell printouts with those exceptions detailed below.

Figure 6-91 illustrates the heading information on the flight test narrow band analysis. The information in the first 3 lines is the same as that of the test cell except that the speeds listed are corrected speeds. For the flight test analyses, the actual speeds were corrected to their equivalent 100% rpm value and the data was shifted in frequency a corresponding amount. Line 4 contains the corrected N1 and N2 speeds of the engine, the particular flight condition being analyzed, the aircraft serial number and the aircraft speed. Line 5 is similar to the test cell and line 6 gives the actual rpm values before correction. The data printout format is similar to that of the test cell data.

Except for the mean identification number, the flight test mean and standard deviation printout is the same as that of the test cell. Figure 6-92 is an example of a flight test mean and standard deviation printout. For the flight test the mean identification numbers and their corresponding parameters were as follows.

6.5.4 Continued

Parameter No.	Flight Test Mean Identification No.
7	MNA007
4	MNA004
8	MNA008
45	MN0045
47	MN0047
49	MN0049
123	MN0123
125	MN0125
126	MNA126
129	MN0129
59	MNA059
61	MNA061
64	MNA064
66	MNA066

Figure 6-93 is an example of the mean plus ten standard deviation comparison summary for parameter number 66 "output quill bearings" on the 90° gearbox. This figure illustrates part of the defective part testing during the flight test phase. The flight test comparisons were not sorted according to type of defect as was done with the test cell data because of the small number of defective parts tested. Each run number of Figure 6-90 therefore, is for a different type of defect. Correlation between run number

6.5.4 Continued

and type of defect can be determined by reference to Figures 6-94, 95, 96, and 97 discussed in Section 6.5.6.

In the flight test comparisons, the individual frequency band numbers were printed out for the seven highest comparison levels. While the general format of the flight test comparisons is the same as that of the ground test, the large number of levels which had the "bands greater than" printed out, necessitated simplifying this portion of the printout. For the flight test, only those bands which exceeded one level but dropped out before the next highest level were printed. To determine the frequency bands that exceeded a specific level, look at that level and all levels greater than that level. Referring to Figure 6-93, run 167 for a moment, the comparison summary indicates that three individual frequency bands exceeded the mean plus five standard deviation level. Under "bands greater than" the mean plus five standard deviations there are two band numbers listed: 83 and 422; under the mean plus six standard deviation one band number 47. This means that band numbers 83, 422 exceeded the mean +5S but dropped out before the mean +6S level; band number 47 exceeded the mean +6S level and dropped out before the mean +7S level.

6.5.5 Summary of Flight Test Analysis

6.5.5.1 General

As has been indicated, analysis of flight test data was complicated compared to analysis of test cell data because of various causes. These may be summarized as follows:

1. Small sample size for determination of LRU baselines (i.e., only two transmissions established the baseline).
2. Method of speed compensation left undesirable granularity in band placement, and difficult correlation between N1 and N2 effects.
3. Transmissibility in aircraft made isolation of "cause-effect" relationship more difficult than anticipated.
4. The degree of "badness" was in general unknown.

The net result of the first two problems was that the coefficient of variance for the different bands was significantly higher than desirable. Therefore, many significant changes in the vibration signature might not cause 4σ exceedances.

The result of the third problem is to increase the number of 4σ exceedances for a particular LRU due to coupling of energy between LRU's.

The result of the fourth problem is a difficulty in distinguishing between good and bad parts.

Realizing these problems exist and also realizing that a decision had to be made despite these problems, the scope of the analytical technique previously used for test cell analysis was widened to try to obviate these shortcomings.

6.5.5.2 Problem Analysis Technique

Nothing could be done in an analysis technique to cope with problem (1) which was clearly a question of an insufficient statistical base.

The data could not be reprocessed in time to fully compensate for problem (2). In order to partially compensate, however, a $\pm$ 1 band inaccuracy was assumed in the Fourier bands.

The fact that fewer than hoped for 4 σ exceedances resulted from problems (1) and (2) could not reasonably be adjusted for without extensive data re-analysis; therefore, this was simply recognized as a problem which would limit the effectiveness of flagging a bad unit.

The transmissibility effect was corrected by the following procedure:

a. The expected bands due to gear mesh and side bands of each mesh were calculated for pertinent sensor location.

b. The expected bands due to rotational speed and harmonics were calculated for each sensor location.

c. The bands were then compared to determine what bands were common between sensors, and what bands were unique.

d. Comparison summaries were then prepared for each known bad parts flight. A comparision summary lists all band exceedances for each transducer which exceeds the mean plus 4 σ, where the mean and σ for the various bands is as determined from the known good parts flights.

e. The 4 σ exceedances for each sensor were then studied to see if they corresponded to an expected band of interest. If they did, they were weighted 2 or 1, depending upon whether they were unique or common bands respectively. If they did not, the exceedance was rated as zero. The

6.5.5.2 e. Continued

concept of utilizing a weighting factor was utilized in order to provide a better means of separating good and bad parts. In order to demonstrate the usefulness of weighting the bands, the above-mentioned rating factors were chosen. Later the weighting factors were expanded in an effort to further improve the "good, bad" part discriminations. This is elaborated upon in Section 10, Paragraph 10.4.4.1. The effectiveness of weighting factors was clearly demonstrated in the improved ability to separate good and bard parts (i.e., make their spread in ratings greater, reference Paragraph 10.4.4.1). The next step would be to provide a computer with all the data (i.e., 4σ exceedances for good and bad parts plus the good-bad parts definition) and allow it to optimize the weighting factors for the different bands. The computer program was, however, beyond the scope of the program.

f. All sensor exceedan[illegible]re similarily computed and all sensor outputs on an IRU were summed.

Problem (4) then became evident; i.e., the degree of badness in the parts installed. The spread on the answers due to the above approach made choice of a good/bad level difficult. It should be emphasized, however, that had problems (1) and (2) not existed, more 4σ exceedances would be expected in the bands of interest for the bad parts and, therefore, much improve the sensitivity of our analysis to discriminate between good and bad parts. Phase B will study the improvement brought about by resolving problem (2).

6.5.5.3 Derived Evaluation Criteria from Known Bad Parts Tests

From the known bad parts testing, the following levels were chosen as bet fits to detect the known bad parts:

Engine	7
42° Gearbox	4
90° Gearbox	12
Transmission	22

Sensors 45 and 49 on the transmission were not analyzed nor was sensor 59 on the 42° gearbox. At the time these were felt redundant.

The engine rating of 7 was correct in 15 out of 20 cases, with 2 being borderline cases (i.e., an engine called good with a rating of 6, and bad with a rating of 9).

The 42° gearbox rating of 4 was right in 17 out of 20 cases.

The 90° gearbox rating of 12 was right in 14 out of 20 cases.

The transmission rating of 22 was right in 13 out of 19 cases (in one case sensors 47 and 129 were not functioning, so the case was thrown out) with 3 borderline cases.

While it is recognized that the score is not perfect, it is strongly felt that considerable improvement is possible by employing data processing and technique refinement (Reference Section 10, Paragraph 10.4.4.1) which are now evident.

6.5.6 Summary Flight Test Data

Figure 6-94, 95, 96, and 97 contain a summary of the flight test data for the level flight test condition on the engine, tranmission, 42° and 90° gearbox, respectively. The information contained in these figures is presented in the same format as that for the test cell data described earlier in paragraph 6.4.6.

SECTION 7

VERIFICATION TEST ANALYSIS

7.0 VERIFICATION TEST ANALYSIS

7.1 Gas Path Data Analysis

The AIDAPS Test Bed Program included a series of technique verification tests. The objective of the verification tests was to identify degraded parts which were implanted in six engines without prior knowledge as to what part had been changed. This section covers the application of the gas path technique to the six unknown engines in diagnosing possible degraded gas producer (compressor, N_1 and N_2 turbine and nozzles, etc.) faults.

7.1.1 General Comments

The six verification engines had not been previously encountered in the Test Bed Program and therefore baseline flights were required on each engine to demonstrate the unique Hamilton Standard gas path analytical technique. (Refer to Section 6.2). The data resulting from these tests is displayed in bar chart form in Figures 7-2 through 7-13. These charts depict the change of the measured parameters from a baseline (Figures 7-8 through 7-13) and the engine variations which produced these changes (Figures 7-2 through 7-7). The actual baseline characteristics which were used for the analysis (Figure 7-1) are:

1. The flight of the specific engine in a "good" configuration;
2. The average flight data from the (4) Phase D and (6) verification engines; and
3. A typical engine characteristic extracted from the data provided by Lycoming Model Specification No. 104.33.

A fourth bar is included for two engines, LE 14819 and LE 17376 (Figures 7-4, 7-5, 7-10, and 7-11), which demonstrates the results of an analysis

7.1.1 Continued

based on data tabulated by ARADMAC personnel during the test cell engine checks. This bar illustrates the difference between degraded and rebuilt runs in the test cell. These are the only two engines for which a complete data set is currently available.

The data has been compared to the three separate engine characteristics or baselines listed above in order to demonstrate the potential jeopardy inherent in using the improper baseline. The correct result is the first bar in each parameter group which compares the test and baseline flights for the <u>specific engine</u>. The second bar illustrates the results which would be obtained from an average characteristic based on a small engine sampling. The third bar approximates the results which could be obtained from an average of many engines. Conclusions from this comparison will be summarized following a discussion of the inidividual results.

A final comment involves the limits of degradation which must be encountered before engine maintenance is required. The proper definition of these limits inherently remains the responsibility of the engine manufacturer. The definition of malfunction limits based on the small sampling used in this Test Bed Program would probably prove to be too restrictive for a production system. However, it is proper to define detection limits for the Test Bed Program to distinguish between normal variations and implanted changes. Exceedance of the detection thresholds listed below in terms of percent change allowable indicates a malfunction.

T_{T5}	+5%
W_A	-6%

7.1.1 Continued

η_c	$\pm 3\%$
A_5	$\pm 5\%$
η_T	$\pm 3\%$
A_n	$\pm 5\%$

7.1.2 Verification 1 - LE 18270, Figures 7-2 and 7-8 (July 16, 1971)

The data presented for this engine illustrates that different answers may be obtained from the three analytical approaches*. Application of the limits of detection to the second and third bars (average baselines) results in the diagnostic of No Gas Path Fault. The custom baseline indicates that both the N_1 turbine area (ΔA_5) and the power turbine area (Δ An) have changed by 6% and are thus the probable implants. However, as was mentioned, the baseline flight for this engine experienced a short circuit in the exhaust gas temperature harness. A reduction of the base temperature by 14 degrees (1%) would produce a 2% area reduction for both nozzles and a 2 1/2% airflow reduction. This engine is thus in the gray area in that no change is a definite possibility. The Army Aviation Systems Command, AAVSCOM, confirms that no degraded component had been implanted in this engine.

7.1.3 Verification 2 - LE 20791, Figures 7-3 and 7-9 (July 19, 1971)

This set of engine data indicates that the turbine nozzle areas (ΔA_5 and An) have probably not been changed since all variations are within the limits of detection. This data does illustrate the uncertainty which results from the baseline definition in that the third bar for ΔA_5 is very near a

* An additional complication for this engine was present since for the baseline flight the EGT (T_9) harness had malfunctioned. An extrapolation of ARADMAC test data was used for this parameter.

7.1.3 Continued

detection limit and a slight additional coking would produce an incorrect diagnostic. However, the diagnostic derived for this engine is that compressor airflow (ΔW_A) has deteriorated. Data obtained from AAVSCOM indicates that no deteriorated component had been implanted. Subsequent analysis of the flight data and study of the engine operation capabilities indicates that a change of only 5% in the engine bleed air extraction would change the airflow from -7.2 to -5.2% and result in the diagnostic of No Gas Path Fault.

7.1.4 Verification 3 - LE 14819, Figures 7-4 and 7-10 (July 23, 1971)

This data clearly indicates that a degraded compressor has been implanted in the engine since ΔW_A and $\Delta \eta_c$ have exceeded their detection limits. This conclusion is confirmed by AAVSCOM. The fourth bar of each parameter group which resulted from an analysis of the data on ARADMAC test cell logsheets also confirms this conclusion. The 4% deterioration in power turbine nozzle area (Δ An) might well be reduced significantly had inter-turbine pressure measurements been included in the mathematical model. The Δ An change did not, however, produce an undesired diagnostic message.

7.1.5 Verification 4 - LE 17376, Figures 7-5 and 7-11 (July 26, 1971)

This data indicates that no degraded part was implanted in the engine. The ARADMAC test cell logsheet data (fourth bar) confirms this conclusion. However, AAVSCOM has indicated that the N_1 nozzles were changed for this flight. Subsequent discussions with AAVSCOM and Bell Helicopter representatives have indicated that it is difficult to match the turbine stator (nozzles) and rotor and the apparent disagreement between the data and desired implant could be attributed to this problem. It should be noted that the baseline

7.1.5 Continued

flights occurred after the degraded part flight and the match for this baseline flight may not have been sufficient to permit detection of the implanted component. This test result is thus in the gray area.

7.1.6 Verification 5 - LE 18301, Figures 7-6 and 7-12 (July 30, 1971)

This test indicates that changes have been implanted in the N_1 turbine nozzles and N_2 turbine nozzles because both ΔA_5 and ΔAn exceeded limits of detection. AAVSCOM has confirmed that the N_2 turbine nozzles were changed and thus the diagnostic approach correctly isolated the fault to the turbines. The variation in N_1 nozzle area (ΔA_5) may again be caused by the matching problems on rebuild and compounded by the lack of an inter-turbine pressure measurement. The ARADMAC logsheets for this engine were not available for analysis at publishing time.

7.1.7 Verification 6 - LE 16886, Figures 7-7 and 7-13 (August 12, 1971)

The flight data indicates that no degraded component was implanted for this test because no parameters exceeded their limits. This conclusion was confirmed by AAVSCOM. It is interesting to note that the data for a 10 engine average baseline (second bar) indicates an N_1 nozzle area change (ΔA_5) which is not confirmed by any other data. This deviation again demonstrates that the best confidence can be obtained from the Hamilton Standard approach of comparing a deteriorated engine with the same engine in a "good" configuration.

7.1.8 Summary of Analytical Techniques

The verification test data has been presented for three baseline definitions. In general, the conclusions obtained by comparing the flight

7.1.8 Continued

data to either a 10 engine average baseline or the engine model baseline is not in good agreement with the specific engine results. Further study also indicates that some of the variations, notably in the turbine areas, illustrate opposite signs. This result is quite possible because the degraded flight data can exist between a multi-engine average and the specific "good" engine data. This then confirms the Hamilton Standard approach of only comparing data on a specific engine as yielding the most reliable results.

The gas producer and power turbine areas (ΔA_5 and Δ An, respectively) are actually effective flow areas which are determined by the stator vane and rotor configuration. A positive area change is indicative of either an eroded or worn nozzle or that the angle of attack between stator and rotor is too shallow. A negative area change indicates either a coked or clogged nozzle or a steep angle of attack. Thus it is quite possible for the turbine area changes to be of opposite sign in any given installation.

7.1.9 Gas Path Analysis Restrictions

The above described test results have been successfully formulated in spite of three basic assumptions which tended to restrict the conclusions. First, it must be assumed that the engine characteristics will not change during two disassemblies and reassemblies. The validity of this assumption has not been tested. Second, it must be assumed that the "degraded" parts are significantly different from the original parts. The small variations which were encountered in some instances would indicate that this assumption may not be universally valid. Finally, the "degraded parts tests" utilized two helicopters while the baseline tests used only one helicopter. The effect of this variation has been minimized as much as possible by attempting to retain the sensors with the engine. However, normal wear and attrition has occurred which introduces a degree of uncertainty into the final conclusions.

Considerable improvement of the diagnostic confidence will be obtained in a production AIDAPS system by eliminating the aforementioned restrictions. The initial flights will establish the actual engine baseline characteristics

7.1.9 Continued

for the remainder of the engine life. Component degradation will then be firmly established as wear occurs and the uncertain artificial variations between "good" and "bad" parts will be eliminated. The use of a completely consistent equipment set will eliminate the need to estimate performance differences between helicopter installations, wiring, and sensors. Finally, the effect of the continual engine disassembly and rebuilding process will be eliminated as a potential variation because this process would not occur in a normal application. Any overhaul which is required would result in a new engine baseline characteristic being established concurrent with engine re-installation.

7.2 Mechanical Diagnostic Summary

The entire subject of mechanical diagnostics was discussed in detail in Section 6.3. The conclusion reached was that the degraded parts did not establish a malfunction signature during the flight test phase. This same conclusion is generally applicable to the verification test flights. Those parts installed in aircraft 61011 did not exhibit any significant limit exceedances or diagnostics.

The same conclusion is true of the parts installed in aircraft 17223 with only two exceptions. These involve the bearing temperature rises on the second and sixth engines, LE 20791 and LE 16886, respectively. The oil temperature rises in bearing 2 and bearings 3 and 4 exceeded preset limits for a large percentage of both flights and resulted in the diagnostic to inspect the lubrication system. A further isolation of the actual problem cannot be obtained because this condition was not encountered during the

7.2 Continued

flight test. An attempt to understand this condition involved additional reviews of the data flights on helicopter 17223 as outlined below.

The data flights on AC 17223 during the flight phase all exhibited high bearing temperatures for both bearings with respect to the equivalent temperatures on AC 61011. The bearing 2 temperature rise was consistently near or above the diagnostic limit but no other malfunction indication was present and bearings 3 and 4 temperatures were 40 to 50 degrees below the limit. The verification flights produced bearings 3 and 4 temperature rises which averaged above the limit and bearing 2 had also become higher. This leads to the conclusion that the cooling system on AC 17223 was less efficient than that of AC 61011 and that some change had been incorporated for the demonstration flights of LE 20791 and LE 16886.

7.3 Verification Testing - Vibration

7.3.1 General

The criterion generated in paragraph 6.5.5 was next applied to the verification test results. In these cases, however, sensors 45 and 49 on the transmission and 59 on the 42° gearbox were analyzed. Therefore, good-bad levels had to be established for these components. Based on two sensors being used on the 42° gearbox instead of one, this LRU level was doubled to 8. The transmission sensors 45 and 49 proved to generate a disproportionate number of exceedances, and the limit was, therefore, raised to 100. The verification good-bad levels are summarized below:

Engine	7
42° Gear Box	8
90° Gear Box	12
Transmission	100

The weighted 4 σ analysis described in paragraph 6.5.5 was then calculated with the following numerical rating resulting.

Date	Run No.	Engine Weighted 4 σ	42° Weighted 4 σ	90° Weighted 4 σ	Transmission Weighted 4 σ
16 July	170	6	7	0	145
19 July	173	12	No Data	4	169
23 July	176	13	10	0	47
26 July	179	13	1	5	106
30 July	182	0	10	31	25
2 August	185	16	1	6	183

7.3.2 Further Considerations of Transmissibility Between LRU's

From the results of our known bad parts tests, reference 6.5.5, it was evident that our score could have been significantly improved if certain borderline cases had their decisions reversed. A further look was then taken into the problems associated with transmissibility to see if borderline cases could be swayed one way or the other. Of particular interest is coupling between the engine and the transmission as well as between the 42° and 90° gearboxes.

7.3.2.1 Transmission to Engine

In the case of the engines, run No. 179 and 185 were thought borderline. In these two cases, the transmission had high 4σ exceedances. The eleven bands were studied to see if they could be eliminated because higher energy was present in that band at the transmission than at the engine; this is an indicator of the transmission being the source of this energy rather than the engine. This reduced the weighted 4σ exceedances by 6 and 5 for Runs 179 and 185, respectively. To assure that this approach was indeed improving our overall ability to discriminate between good and bad parts, similar analysis were made for the other verification flights. The results showed an average reduction of less than 3 for the other runs; therefore, producing a net improvement in discrimination. Interaction in the other direction between engine and transmission was not considered for two reasons:

1. High ratio of 4σ transmission exceedance to engine 4σ exceedances.
2. Relatively few overlap bands (less than 8%) between the transmission and engine compared to the total number of transmission bands considered.

7.3.2.2 90° and 42° Gearboxes

The other interactions considered in the final choice involved the 90°-42° gearbox transmissibility. If Runs 170 and 182 are considered marginal on the 42° gearbox, interaction with the 90° gearbox should be considered. Since there were no 90° 4σ exceedances for Run 170, the corresponding 42° rating of 9 was not altered. However, there were 31 90° 4σ exceedances in the case of Run 182. These were then studied to determine what the weighted 4σ would be if the interactive bands were totally dropped, rather than rating them 1. This gave a new weighted 4σ of 6.

None of the 90° boxes were considered marginal, therefore, these were not studied further.

As a result of the above judgment decisions, the final weighted 4σ summary of the verification tests is presented.

Run No.	Engine Weighted 4σ	42° Weighted 4σ	90° Weighted 4σ	Transmission Weighted 4σ
170	5	7	0	145
173	8	0	4	169
176	10	10	0	45
179	7	1	5	106
182	0	6	31	25
185	11	1	6	183

7.3.3 Scoring System

A rating technique was devised as follows to account for the degree of wrongness in the answer, while not accentuating the degree of rightness. The rules adhered to were as follows:

7.3.3 Continued

1. If answer was right - score 10.

2. If answer was wrong - score rating over threshold level times 10 for numbers less than the threshold, and the threshold divided by the rating times 10 for numbers greater than the threshold.

3. The total was divided by 230. (Note 23 X 10 is maximum score, since on Run 173 sensor 61 was not functional; therefore, it was felt unfair to call answer either right or wrong to obtain percent ratio.)

4. Finally, the answer was modified in the case where the gas path correctly called an engine bad if the vibration called it good. In this case the answer was scored 10.

5. G = Good B = Bad

The case of the engine will be worked out in detail as a sample calculation.

Run No.	Engine Weighted 4σ	Threshold	HSD Answer	AVSCOM Answer	Score
170	5	7	G	B	5/7 x 10
173	8	7	B	B	10
176	10	7	B	B	10
179	7	7	B	B	10
182	0	7	G	B	0/7 x 10
185	11	7	B	B	10

7.3.3 Continued

The net effect of inclusion of the gas path is shown below:

Run No.	HSD Composite Answer	AVSCOM Answer	Score
170	B	B	10
173	B	B	10
176	B	B	10
179	B	B	10
182	B	B	10
185	B	B	10

The same procedure is worked out for the remaining LRU's, with the results as tabulated.

Flight	Engine Scores Without Gaspath	Engine Scores With Gaspath	42°	90°	Transmission
170	5/7 X 10	10	7/8	10	100/145 X 10
173	10	10	Delete	10	10
176	10	10	10	10	10
179	10	10	1/8 X 10	10	100/106 X 10
182	0	10	10	10	10
185	10	10	1/8 X 10	10	100/183 X 10
TOTAL	47	60	31	60	52

Grand Total (Without Gaspath) 190/230 = 83%

Grand Total (With Gaspath) 203/230 = 88%

7.3.4 CONCLUSIONS

1. Statistical techniques can be used to sort good-bad parts for isolated LRU testing. A mean spectrum could be generated, and therefore allow a sorting of units in terms of deviation from the mean.

2. Significant amounts of transmissibility have been observed between the various transducers on the transmission and between the 42^{o} gearbox and 90^{o} gearbox. This causes the purely statistical approach to have questionable application unless transmissibility is accounted for.

3. Frequencies associated with pits on bearings have been identified on the 42^{o} gearbox, but the amplitude levels seen to date have been small. A better means of detecting faulty bearings appears to be in the variation they generate in the gear mesh frequency spectrum. In the case of the engine bearings, however, the bearing frequency components associated with pits are much higher in level, and detection is feasible with reasonable analysis techniques.

4. All measurements should be speed corrected, in the case of the engine for example, two outputs should be generated, one speed corrected with respect to N1, one with respect to N2 to allow good discrimination between close gear mesh ratios, bearing frequencies and sideband structure. Also a finer analysis bandwidth would allow more accurate isolation of frequencies associated with a malfunction.

5. The modulation process that accounts for the extensive sideband structures associated with gears and bearings is predominantly frequency modulation. This has several important ramifications concerning the extension and refinement of the diagnostic techniques developed on this program, and which are still being investigated.

6. Transducer location is critical. In general, transducers on the power train LRU's were located as close to the expected source of vibration as possible. However, the engine transducers were less optimally located. While it is recognized that it is characteristically difficult to mount accelerometers close to actual engine bearings, this is highly desirable. Additionally, care should be taken that no structural resonances are present in the surface the accelerometer is mounted upon. In the case of the AIDAPS installation, some compromise had to be made in both of the above requirements to provide expedient installation of the accelerometers.

7. A statistical approach modified by weighting factors has resulted in the ability to detect unknown implants with a success in excess of 80%. Further refinements are expected to improve this number. The approach utilized would lend itself to in flight implementation with reasonable hardware.

7.4 Standardized Scoring Summary

This section will present the results of Phase D verification tests according to the following restrictions as set down by AVSCOM:

1) Any component which was in fact a marginal part must be considered to be bad for scoring purposes.
2) The score for each component must be expressed in straight percentage form; i.e., the number of correct answers out of six.
3) The agregate score for all component runs must be expressed in straight percentage form; i.e., the number of correct answers out of 24.

In addition, a general commentary on this scoring technique will be included.

7.4.1 Engines

Out of six runs Hamilton Standard diagnosed the condition of four engines correctly for a score of 67%.

7.4.2 Transmissions

Out of six runs Hamilton Standard diagnosed the condition of four transmissions correctly for a score of 67%.

7.4.3 42° Gearboxes

Out of six runs, Hamilton Standard diagnosed the condition of three gearboxes correctly for a score of 50%.

7.4.4 90° Gearboxes

Out of six runs, Hamilton Standard diagnosed the condition of four gearboxes correctly for a score of 67%.

7.4.5 Composite Score

Out of the 24 combinational component-runs Hamilton Standard diagnosed the condition of 15 components correctly, for a composite score of 63%.

7.5 Commentary

As has been previously stated, the technique of implanting bad parts in components has some significant drawbacks associated with it which causes such an experiment to suffer from a lack of control. Specifically, the degree of badness of the parts makes absolute diagnosis marginal. In addition, the effect of disassembly and reassembly to accomplish implantation puts an abnormal strain on the gas path technique of continuous comparison to a customized baseline. Further, the statistical sample size for vibration evaluation was small. (Two units in the case of transmissions.)

If in fact the scoring is re-examined in a manner which deletes the parts which are marginal, the scores will alter as follows:

Examine only those components which were good. (i.e., transmissions from Runs 1, 3, 4, 5, and 6; 42° gearbox from Run 5; 90° gearbox from Runs 2, 3, 4, and 5) Out of a total of 10 good components, 7 were identified correctly by Hamilton Standard as good for a score of 70%.

Examine only those components which were truly bad (i.e., 42° gearboxes from Runs 1, 2 and 3; 90° gearbox from Run 5; and Engines from Runs 1, 2, and 3.)

Out of a total of 7 bad components Hamilton Standard correctly identified 6 as being bad for a score of 86%.

The total combinational score considering no marginal parts but only good or bad components is 13 out of 17 or a score of 76%.

Seven of the parts used in the test were adjudged by AVSCOM as marginal. Instead of arbitrarily calling these marginal parts bad, assume that these fall into a "gray area" of the good-bad demarcation. That is, allow a tolerance around the criteria judgement as to good or bad parts. Considering this, then Hamilton Standard correctly identified 20 out of the 24 component runs for a score of 83%.

SECTION 8.0

CONCLUSIONS

8.0 CONCLUSIONS

The conclusions listed in this section are all extracted from the results detailed in Sections 5-7 and Section 10. These conclusions are listed below together with appropriate section source references.

8.1 General

- AIDAPS state-of-the-art equipment functioned successfully in the Army helicopter environment. (6.0, 7.0, 10.0)
- There was no interference in helicopter system functioning due to the instrumentation and on-board electronics equipment. Conversely, there was no adverse effect on AIDAPS from the helicopter systems. (3.8.4, 3.8.5)
- The degree of component operational badness was found to vary from very slightly worn to field replaceable bad in Phases B and D. This implies that parts judged to be mechanically bad by conventional standards can still operate properly.
- Components selected for Phase E worse parts testing were deliverately chosen to display further degradation. This result was indicated quite positively by AIDAPS (10.4).
- Analysis of the Bell Helicopter Company rotor tracking and balance technique shows it is quite feasible for implementation within AIDAPS. (10.3.3)
- Diagnostic software did require some modification to effectively utilize hover data as contrasted with level flight. Modifications involved determination of stable gas path operating points for engine analysis and adjustment of vibration thresholds to recognize hover conditions. (10.4.4.1, 10.4.2.3)

8.1 Continued

• Using the combined methods of mechanical, Gas Path, and vibration diagnostics resulted in isolation to LRU effectivity of approximately 88% for Phase D verification tests. Score is not known for Phase E verification. (7.3.3)

• Inspection capabilities of AIDAPS were demonstrated by the lack of non-existent problems flagged. Although the parameter limit checking was not deliberately exercised, results were consistent with actual aircraft inspections and performance. (6.3, 7.2)

• Trending was carried out for approximately 250 hours on numerous parameters. This was not large enough to generate any extreme trends but did show some definite patterns. (5.3.5, 6.2.3.3.2, 6.2.3.3.3)

8.2 Engine Diagnostics

• Hamilton Standard Gas Path Analysis was readily implemented for the existing T53-L13 engine. Sensors were all conventional and mainly consisted of standard engine instrumentation. (6.2)

• Meaningful engine operating thermodynamic information was acquired from only five engine parameter measurements. (6.2)

• Gas Path Analysis was successfull in fault isolation not only to the LRU but also to the component level. (7.1.2-7.1.7)

• Lycoming "hot end" analysis technique was studied but not implemented within AIDAPS because of the relatively few exposures experienced compared to those required to achieve the threshold indices. (5.3.3.3)

8.2 Continued

• Custom baselines should definitely be utilized to enhance Gas Path accuracy because of the variations between engines. Gas Path Analysis signatures were confirmed between test cell and flight test with known bad parts. Gas Path analysis thresholds were successfully implemented to distinguish between good and bad parts. (6.2.3.2.4, 7.1.1)

• Engine diagnostics utilized combination of Gas Path Analysis and detection of non gas path problems through vibration and mechanical diagnostics. Vibration amplitudes were readily distinguishable for engine bearings because of the high rotational speeds and centrifugal forces, but needed correction for extraneous other bearing frequencies and speed. There were no deliberate mechanical diagnostic implants but mechanical diagnostics did perform properly. (7.2. 6.4.5, 5.3.3.1)

8.3 Gearbox and Transmission Diagnostics

• The principle technique utilized was statistical vibration analysis. Mechanical limit checking was also utilized through tests such as chip detection, oil temperature, and oil pressure but were not sensitive enough to detect artificially implanted problems during flight test times. (6.0)

• A very thorough analysis of vibration detection in the helicopter environment was performed during Phases C and D and is completely documented in Sections 6.3 through 6.5. (6.4.3.3)

8.3 Continued

- The vibration analysis technique established is based upon digitizing the analog data and performing a Fast Fourier Transform (FFT). A statistical comparison of the spectral energy levels as compared to good parts signatures enables statistical discrimination of bad parts (6.4.3.4).

- This method was extensively tested during Phase B and was completely successful in differentiating good LRU's from bad LRU's.(6.4.5).

- Flight testing indicated refinements of the vibration technique where necessary to correct for transmissability effects, aerodynamic noise, and the structural mounting of LRU's (6.5.3).

- Adjustment for the above effects involved evaluation of the statistical distribution utilized, the spectrum width (the number of power spectral lines computed) transmissability (amount of energy transfer between LRU's) modulation effects (the effects of faults on power spectral distribution) reactions (the effects of bearing faults on gear mesh energy) and mechanical properties of gears, bearings, and shafts (6.5.3.4 - 6.5.3.6).

- Vibration analysis technique was successfully implemented for flight tests as a result of the above studies. Main factors involved an understanding of the gear clash sidebands and harmonics together with the frequency modulation (FM) process resulting from the interaction of faulty bearings, shaft eccentricity, and gear mating abnormalities.(6.5.3.6, 6.5.5).

8.3 Continued

. Gear clash frequency bands are most predominant in energy level. Bearings are directly observable for transmissions and gearboxes but are of a lower energy level. Successful diagnostics to the LRU level are feasibly accomplished by analysis of the extensive gear mesh frequency spectrum. (7.3.4)

. FM effects are a very sensitive indicator of shaft gear or bearing problems including tolerances, wear, and dynamic loading. Analysis of the extent of the sideband structure and energy distribution can isolate problems to the component level. (6.5.3.6)

. Second and third harmonic energy distributions are different from the fundamental. (6.5.3.6)

. Transmission testing to destruction at Bell Helicopter Company resulted in a failure pattern for a bad part which detected operational failure 40 hours in advance. (10.5.3)

. There are strong indications that vibration trending can be accomplished by repeated spectral analysis to ascertain shifts in mechanical operation. (6.5.3.6)

. Statistical analysis techniques were sensitive enough to isolate degrees of badness of parts in terms of supposedly good parts used for reference versus the judged bad parts. This leads to a re-evaluation of the operating capability of parts conventionally judged degraded by common inspection means. (6.5.3.6)

8.3 Continued

- Significant amounts of transmissibility were observed between the various transducers on the transmission and the 42° gearbox and the 90° gearbox. The purely statistical approach must be corrected for this effect. The modulation process accounting for the extensive sideband structures associated with gears and bearings is predominantly frequency modulation. The vibration technique developed has been refined and verified through the Test Bed Program. The approach utilized is feasible for in-flight implementation. (7.3.4, 10.4.4.1)

SECTION 9.0

RECOMMENDATIONS

9.0 RECOMMENDATIONS

- Engineering service test should be performed for a suitable length of time with enough aircraft to generate a good trend data base and to expand the data base used for vibration threshold determination. This test will also enable evaluation of the impact of AIDAPS in the normal aircraft operational environment. It is strongly felt that AIDAPS will show its real potential effectivity under these conditions because of the expanded data base, consistency of baselines, and avoidance of undue AIDAPS equipment changes.

- Work should be done on torque and fuel flow sensors to improve repeatability. Consideration should be given to development of a real-time oil analysis transducer for use with AIDAPS.

- Vibration measurements should be speed corrected to allow discrimination between close gear mesh ratios, bearing frequencies and sideband structures. Transducer vibration locations should be as close to the expected source of vibration as possible. In particular, engine transducers should be more optimally located. The engine gearbox should be instrumented with vibration transducers for further engine vibration testing.

- Rotor tracking and balance calculations should be included within AIDAPS.

- Service test data should be utilized in the vibration analysis program to further refine generation of means and confirm fault isolation to the component level.

9.0 Continued

- The vibration technique developed is feasible for airborne implementation.

- The airborne AIDAPS may well be utilized with a time shared ground computer to form a Hybrid System where the ground computer serves such functions as hard copy printouts, extended trend predictions, and generation of logistics records.

- Bad parts testing during AIDAPS should be supplemented by further tests to establish a range for degrees of badness between marginally bad and not safe for operation.

APPENDIX I

SECTION 10

PHASE E PROGRAM

10.0 PHASE E PROGRAM

10.1 Introduction

Based upon the promising results of Phases A through D, the Army chose to extend the UH-1 AIDAPS program to selectively pursue certain areas of special interest. These areas were divided into three tasks: (I) aircraft (UH-1H) main rotor and tail rotor out-of-balance and out-of-track diagnosis, (II) additional degraded component flight testing, and (III) analysis of bearings in final failure modes. Testing was carried out at Army Aeronautical Depot Maintenance Center (ARADMAC) at Corpus Christi, Texas and the Bell Helicopter Company in Fort Worth, Texas.

Phase E commenced on October 7, 1971, following the completion of Phase D additional baseline flights. Testing was substantially completed by January 1972. Phase E and the present UH-1 Test Bed Program will culminate on approximately March 30, 1972, with the completion of all data analysis, report preparation, and contract data item submissions. The following paragraphs constituting Section 10 detail Hamilton Standard's Phase E activities as keyed to the three (3) tasks listed above.

10.2 Scope of Work

10.2.1 Task I. Aircraft (UH-1H) Main Rotor and Tail Rotor Out-of-Balance and Out-of-Track Diagnosis.

Very little data was gathered during the Phase D flight portion of the Test Bed Program regarding the out-of-track (OOT) or out-of-balance (OOB) conditions of the main and tail rotors. The limited data base together with the lack of adequate time during Phase D to properly establish the characteristics of OOT and OOB rotors prevented formulation of a rotor monitoring technique.

Thus, it was a Phase E objective to determine if field adjustments can be predicted based on "as is" measurements to minimize one-per-revolution main

10.2.1 Continued

and tail rotor vibrations. The AIDAPS was employed to measure and record pylon motion, vibration amplitudes, and phase angles during various flight conditions with pre-adjusted known rotor characteristics. From this data, BHC has developed an analysis program to predict optimum rotor adjustments which has been evaluated by Hamilton Standard for feasibility of integration into AIDAPS. Section 10.3 discusses the Hamilton Standard data gathering and implementation efforts.

10.2.2 Task II. Additional Degraded Component Testing.

At the close of Phase D, it was determined that some of the known bad implanted degraded components were actually only marginally bad in terms of operating capability. This was not desirable since it complicated the positive evaluation of AIDAPS effectivity through controlled experiments.

Consequently, during Phase E, the same AIDAPS previously utilized was tested as to effectiveness in detecting, fault isolating, and predicting malfunctions in the UH-1H aircraft using degraded components (engine, transmission, 42° and 90° gearboxes) which were judged in worse condition than those previously tested in Phases B and D. The same Hamilton Standard developed diagnostic techniques were employed with results as described in Section 10.4.

10.2.3 Task III. Analysis of Bearings in Final Failure Modes.

The purpose of this task was to gain additional diagnostic insight into the failure mechanism of transmission bearings through identifying progressive failure rates for known initial degraded conditions and evaluating these rates for feasibility of detection and prediction. A transmission test cell at BHC,

10.2.3 Continued

Ft. Worth was equipped with suitable s[illegible]rs, wiring harnesses, and vibration instrumentation by Hamilton Standard for the conduct of the tests. Analysis of the acquired data is presented in Section 10.5.

10.3 Rotor Unbalance Measurement and Implementation Concept

10.3.1 Test Conduct

The UH-1H helicopter was instrumented to record pylonmotionmain and tail rotor one-per-rev vibrations and rotor azimuth. Data was recorded at a given baseline condition. Flight data was recorded on four main rotors and three tail rotors. For each main rotor, three stepwise adjustments were made on four parameters: roll (pitch link length), tab (blade tab angle) sweep (drag brace), and balance (span wise). For each tail rotor, three stepwise adjustments were made on three parameters: roll (pitch link length), chord balance, and span balance.

Following each adjustment, a specified standard flight profile was flown to collect a minimum of ten seconds of stabilized data at each flight condition in the profile. The data was forwarded to BHC for development of a suitable adjustment prediction program. This analysis was forwarded by BHC to Hamilton Standard where a concept for implementation within the on-board diagnostic system (AIDAPS) was evolved as discussed below.

10.3.2 Rotor Unbalance Test Measurement

The tests conducted were designed to monitor and record the once per revolution vibrations of the main and tail rotor so as to gain the knowledge necessary for developing a suitable analysis technique which will permit field adjustments to be made to minimize the once per revolution vibrations. Figure 10-1

10.3.2 Continued

is a list of the main and tail rotor parameters measured, their once per revolution frequency, and their expected range of amplitudes. Figure 10-2 lists the type transducers used to make the measurements, the transducer manufacturer, the model number, the serial number, the calibrated sensitivity, and the tape track on which the transducer signal was recorded.

In addition, Figure 10-2 also lists the type of signal conditioner and amplifier used with each transducer and the equivalent Full Scale Standardize quantity in transducer units. Since the amplifier attenuator settings are automatically set to X1 in the Full Scale Standardize mode, the equivalent full scale quantity during data acquisition is obtained by multiplying the Full Scale Standardize quantity by the particular amplifier attenuator setting used during data acquisition as indicated by Figure 10-2.

Figure 10-3 is a signal block diagram of the data acquisition and recording system employed on the rotor unbalance test program. Transducer signals are routed to a signal conditioner, these to an amplifier, and finally to an AMPEX AR-200 magnetic tape recorder.

The signal conditioner used on the rotor unbalance program provided three functions:

(1) Routed the proper excitation voltages to those transducers requiring excitation.

(2) Housed a band pass filter which allowed only the once per revolution component of the total transducer signal to be recorded. The center frequencies of the band pass filters used were 5.4 Hz and 27.6 Hz.

10.3.2 Continued

The 5.4 Hz filter was a Burr Brown 5719-BP2N-5R40 and the 27.6 Hz filter was a Burr Brown 5719-BP2N-27R6.

3. Allowed calibration signals to be routed to the input of the band pass filters. The calibration signals were symmetrical square waves of precise amplitude at frequencies of 5.4 Hz and 27.6 Hz. The calibration signals were derived from precision stable oscillators and associated circuits built into the recording system for these tests. The amplitude of the square wave calibration signals was adjusted so that when the Full Scale Standardize mode of the recording system was selected the output of the signal conditioner would be a sine wave of 10 peak millivolts amplitude. This signal was then routed to an amplifier.

The amplifiers used to increase the transducer signal levels had frequency response from DC to 20 KHz. This wide bandwidth avoided consideration of any phase shift the transducer signals might experience in passing through the amplifier circuitry. The amplifier specifications are listed below:

1. Input Impedance — 10 megohms (differential)
2. Output Impedance — 0.1 ohm
3. Gain — 47 db (250) (Full scale input is $\pm$10 mv at an attenuator setting X1)
4. Gain is adjustable by attenuator in steps of 1, 2, 5, 10, 20, 50. The attenuator is automatically set to X1 during the standardize modes.

10.3.2 Continued

5.	Frequency Response	DC to 20 KHz ±10%
6.	Dynamic Range	50 db over full bandwidth
7.	Linearity	±.01%
8.	Common Mode Rejection	110 db DC to 60 Hz
9.	Zero Drift	±.02% Full Scale in 200 hrs.
		±.001% Full Scale/°C change

The signals from the DC amplifiers were then routed to an AR-200 tape recording system. The signals were recorded using an FM record technique at 15 IPS. The recording head assemblies used on this program were configured to Ampex Standard specifications. Figure 10-4 indicates the two seven track head assemblies and the location of the various tracks with respect to the magnetic tape. Analysis of the recorded rotor unbalance data was performed by the Bell Helicopter Company.

10.3.3 Rotor Unbalance Integration Concept

10.3.3.1 General Comments

As a result of the testing at Bell Helicopter Company, it has been concluded that a rotor balance system can be implemented with a pilot's seat vertical accelerometer, a swash plate forward/aft accelerometer, and a pulse pickup for azimuth reference. The output of the first accelerometer will allow aerodynamic balance while the second will allow dynamic balance.

The following paragraphs outline the recommended hardware and software implementation of the system.

10.3.3.2 Hardware Implementation

In order to implement the system, (which will be further elaborated upon in Section 10.3.3.3) X and Y resolved components of the pilot's seat vertical acceleration and swash plate forward/aft acceleration are required where the X and Y coordinates are based on a phase reference derived from an azimuth pulse pickoff. Figure 10-5 displays in block diagrams form the conditioning of the sensor inputs.

Each accelerometer will be first preconditioned by a buffer amplifier to convert the acceleration into a voltage signal. This signal will then be passed through a 5.4 Hz pass band filter to attenuate signals other than the one per rotor rev. signal. The azimuth pulse pickup is first pulse shaped, and then fed directly to two balanced modulators as well as a 90° phase shifter. The quadrature output of this phase shifter will also feed two balanced modulators. The modulators are fed, as shown in Figure 10-5, from the output of the two above mentioned 5.4 Hz filters. Each balanced modulator is followed by a low pass filter to attenuate the 5.4 Hz ripple. The outputs achieved at X_1, X_2, X_3, X_4 are thus as summarized below:

X_1 = pilots seat vert. accel X sin θ,

X_2 = pilots seat vert accel X cos θ,

X_3 = swash plate forward/aft acceleration X sin θ,

X_4 = swash plate forward/aft acceleration X cos θ.

X_5 is simply a DC voltage proportional to the rotor speed so that phase error introduced into the system due to the rotor frequency being not exactly at 5.4 Hz can be compensated for if desired. At the present time, this correction

10.3.3.2 Continued

is not felt necessary. The five inputs X_1 through X_5 are then fed to a multiplexer and from there into the A/D multiplexer all steps being under computer control so that the data can be properly digitized and read into the computer.

Hardware will also be provided to drive various MAAP indicators, under computer control, to indicate adjustments required to aerodynamically and dynamically balance the rotor system.

10.3.3.3 Software Implementation

Balancing or "sweetening" of the rotor involves an aerodynamic as well as a dynamic balance adjustment. Two adjustments, known as roll and tab, are utilized for aerodynamic balance. Similarly, two adjustments, balance and sweep, are used for dynamic balance. Pilot's seat vertical acceleration is used to detect aerodynamic balance, while dynamic balance is determined by forward/aft acceleration of the swash plate.

It has been found, by testing incremental changes of these four parameters, that the following facts appear to be true in general.

1. Balance and sweep adjustments leave little effect on aerodynamic balance (as monitored by pilot seal vertical g).
2. Roll and tab effect aerodynamic as well as dynamic balance.
3. The pilot's seat g vectors introduced by roll and tab adjustments are roughly at the same azimuth angle and vary linearly (or nearly so) with adjustment increment.
4. Tab adjustments seem to be useful in compensating for velocity sensitive aerodynamic unbalance, while roll adjustments are better to compensate for non-velocity sensitive aerodynamic unbalance.

10.3.3.3 Continued

5. The swash plate acceleration vector introduced by balance adjustments is approximately in quadrature to the swash plate acceleration vector introduced by sweep adjustments.

Based on preliminary investigation, it is estimated that 6 plus and 6 minus adjustments for each of the four variables (roll, tab, balance, and sweep) would allow the majority of rotors to be sweetened. (Note plus and minus denote in phase and 180° out of phase vector, respectively. From the above information, the following implementation of the software program is suggested.

Resolved components (X and Y) for the pilot's seat vertical acceleration and swash plate forward/aft acceleration would be stored for 6 positive and 6 negative increments of roll and tab adjustments based on average flight test baselines. Storing the resolved components allows the ability to easily store both magnitude and phase of these vectors. Similarly resolved components of swash plate forward/aft acceleration would be stored for 6 positive and 6 negative increments of balance and sweep. The following data, acquired from flight test, would then be utilized:

1. Resolved pilot's seat vertical acceleration at 80 and 120 knots,
2. Resolved swash plate forward/aft acceleration at 80 and 120 knots.

The resolved pilot's seat acceleration at 80 knots would be used in conjunction with the roll adjustment table mentioned before to minimize the force vector. This would be accomplished by solving the following equation:

$$\arc \tan \frac{y_1 - y_2}{x_1 - x_2} = \theta_1$$

10.3.3.3 Continued

(where x_1 and y_1 are unbalance rotor components, and x_2 and y_2 are force vector components for a given roll adjustment). Repeated comparison of this θ_1 value against a stored θ value for that particular roll adjustment in an iterative manner enables $\Delta\theta$ error to be minimized.

The roll adjustment would then be added to the pilot seat input at 120 knots. Next the tab adjustment table would be utiized by again solving the above equation for θ_2, comparing this θ_2 against a stored value, and repeating at various tab table entries until θ_2 is minimized.

This would complete the aerodynamic balance phase. The final step would be to provide dynamic balance. This would be done by modifying the swash plate forward/aft acceleration at 80 knots per the roll and tab adjustments made, using the stored average flight test baselines mentioned previously. The X and Y components of this new vector would then be reduced to as near zero as possible by the iterative addition of balance and sweep adjustment vectors per the stored average flight test baselines. This is possible since the balance and sweep vector are nearly in quadrature, therefore, the X and Y vectors can both be minimized.

10.3.3.4 Impact on AIDAPS by Addition of Rotor Balance Capability

It is estimated that the above software capability, including table storage, can be implemented in less than 250 words of memory. Hardware required to interface with the computer and MAAP should represent approximately 3, 1/2 ATR size-printed circuit boards of circuitry.

10.4 Worse Degraded Component Flight Tests

10.4.1 Test Conduct

Unknown worse degraded parts were implanted in the UH-1H aircraft for flight testing, analysis, and evaluation by the Hamilton Standard AIDAPS. Two flights each were made with eight sets of possible multiple faults in the engine, transmission, and 42° and 90° gearboxes. A pair of flights with a single set of faults was performed on each of the dates:

	FLIGHT DATE	ENGINE
1.	November 19, 1971	LE 18270
2.	November 24, 1971	LE 17376
3.	December 3, 1971	LE 14819
4.	December 7, 1971	LE 14819
5.	December 9, 1971	LE 14819
6.	December 10, 1971	LE 14819
7.	December 14, 1971	LE 14819
8.	December 16, 1971	LE 14819

In each case, the aircraft flight profile was limited to flight idle (on the ground), intermediate power (light on skids), and hover (within ground effects). The same mechanical analysis, Gas Path Analysis, and vibration analysis diagnostic techniques previously implemented and explained were employed. Again, both Gas Path Analysis and vibration methods were used together to establish engine condition. Results as evaluated by Hamilton Standard were quite positive indicating that the worse degreaded components did, indeed, yield more predominant diagnostic signatures. Some revisions to the software were necessary to properly analyze data acquired under hover conditions. The following sections fully explain the data analysis and results. Figure 10-6 tabulates the component

10.4.1 Continued

health results submitted to the Army by Hamilton Standard after the completion of flight testing.

10.4.2 Engine Analytical Conclusions

The available data included three engines in eight configurations. Two hover runs were conducted for each configuration. A detailed analysis was completed for at least one data tape on each of the indicated dates.

10.4.2.1 Mechanical Parameters

No significant limit exceedances were encountered during the Phase E tests. This result was anticipated since these tests were concentrated on the gas path and vibration areas.

10.4.2.2 Gas Path Diagnostics

The data for each engine is presented as bar charts in Figures 10-7 through 10-14 of this report. Figures 10-7 and 10-8 present a summary of the average result (on which the following conclusions are based), and the results obtained from several steady state tests (as outlined later in this report) are presented in Figures 10-9 through 10-14 for the individual engines.

Engine LE 18270 exhibits the characteristics of degraded nozzles for both turbines. The gas producer nozzle area (ΔA_5) is degraded open by 11% and the power turbine nozzle area (ΔA_n) is degraded open by 9.5%. The compressor parameters (ΔW_a and $\Delta \eta c$) have varied only a slight amount which indicates that the compressor was probably not changed.

Engine LE 17376 exhibits the characteristics of degraded gas producer turbine and nozzle. The gas producer nozzle area (ΔA_5) is degraded closed by 4.9% and the turbine efficiency ($\Delta \eta T$) has decreased by 4.9%. The other

10.4.2.2 Continued

parameters do not exhibit a significant variation.

Engine LE 14819 is essentially an engine with no gas path faults. Minor variations were encountered in all parameters but none were sufficient to exceed a detection limit.

10.4.2.3 Engine Steady State Tests

The original steady state tests which were used in the other AIDAPS data analysis were found to be too restrictive for the short hover runs of Phase E. The steady state data obtained from the hover condition was not considered to be sufficiently reliable and additional data conditioning and smoothing tests were incorporated in order to properly isolate steady state operation. These new tests, as outlined below, yielded good correlation with the original tests on a normal AIDAPS flight test. The titles of the six revised steady state tests are:

1. EGT A;
2. EGT B;
3. FLOW A;
4. FLOW B;
5. LAG: and
6. SLOPE.

EGT A: This test is essentially the same as the original steady state test.

EGT B: This test is similar to EGT A with wider tolerances.

FLOW A: This test considers the fuel flow scatter in a frame as an indication of steady state operation.

LAG: This test approximates the thermal conditions of the engine. Each parameter is modified by an exponential decay function and steady state is defined when the difference between the lag output and the instantaneous frame value for <u>all</u> parameters is within the specified limits.

10.4.2.3 Continued

SLOPE. A linear equation was calculated for 10 frames of data (approximately 20 seconds) for each gas path parameter using a least-square error fit. Steady state was then defined when the change in slope between data blocks for all parameters is within the specified limits. The nominal value for the current data block was used in the gas path analysis.

10.4.3 Worse Degraded Parts Vibration Analysis

10.4.3.1 Phase E Vibration Testing

During Phase E additional vibration data was gathered through flight testing. The vibration sensors and recording system were identical to that used during Phase D. Phase E discrepant parts were installed on the aircraft without any knowledge of the particular faulty implant by the contractor. The objectives of this phase were:

1) To obtain vibration signatures on power train bearing components that represented a fault condition worse than that observed during Phase D,

2) To identify in which LRU the faulty bearings were implanted by comparing vibration spectra.

10.4.3.2 Vibration Data Analysis

All vibration data comparisons of Phase E data were made with the helicopter in the hover mode. The data analysis of Phase E used the same statistical approach outlined during Phase D testing. A new mean and standard deviation spectrum was generated for the flight hover condition.

10.4.3.2 Continued

Data on eight verification runs with unknown faulty parts were obtained during Phase E. Figure 10-15 is a tabulation of the various flights by run numbers versus the number of 4 σ exceedances by filter band number for each of the transducers on the helicopter. Transducer No. 66 was not used.

10.4.4 Vibration Analysis Evaluation Technique

10.4.4.1 Refinements in Analysis Technique Based on Further Study of Phase D Data

Prior to analyzing the Phase E data, Phase D data was re-analyzed to try to provide a more straightforward numerical technique (i.e., no judgment decisions required, reference paragraph 7.3.2) and a better match between the numberical ratings and the final AVSCOM answers. The analysis technique outlined in 6.5.5.2 was taken as the basis, and various iterations were tried. The technique that evolved as the best of those tried basically altered sections a and e of paragraph 6.5.5.2. Section a was altered to include the second harmonics of the gear mesh frequency and its sideband structure whenever these bands fell within the analysis region. Section e was altered to place more emphasis on certain bands. The gear clash frequency and the first two upper and lower sidebands were given a X2 rating factor. Also the fundamental and first two harmonics of rotational frequency were similarly rated X2. Band interference was considered as before, even with the added bands, by weighting them 2 or 1, depending upon whether they were unique or common bands respectively. Four sigma exceedances could not therefore be rated either 0, 1, 2, or 4 as indicated in the table below.

10.4.4.1 Continued

DESCRIPTION	RATING
Non-pertinent band	0
Pertinent band but with interference band elsewhere on the aircraft and band not fundamental or one of first 3 sidebands	1
Pertinent band but with interference band elsewhere on the aircraft but band is fundamental or one of first 3 sidebands or pertinent band with no interference band but band is not fundamental or one of first 3 sidebands	2
Pertinent band, no interference band, and band is fundamental or one of first three sidebands	4

The improved technique provides significantly better separation between the good and bad parts. Phase D results on the 42° gearbox are presented below to illustrate this point.

a) Phase D Known

Bad Parts — Based on Threshold of 5, 19 Out of 20 were Right

b) Phase D

Verification :

Date	Flight No.	Original Analysis Technique	Original Analysis With Judgment	Original Threshold	Revised Technique	Revised Threshold	AVSCOM Answers
16 July	170	7	7	8	28	10	B
19 July	173	No Data	No Data	8	No Data	10	-
23 July	176	10	10	8	31	10	B
26 July	179	1	1	8	1	10	G
30 July	182	10	6	8	16	10	G
2 Aug.	185	1	1	8	8	10	G

10.4.4.1 Continued

A much better match is evident with the revised technique, plus no judgment decision is required. The establishment of a final threshold level was also reviewed. This is felt to be a problem that could be eliminated with a larger data base. A level of 10 was chosen as a compromise between the Phase D known bad parts flights and the Phase D verification.

A similar analysis was made for the engine, transmission, and 90° gear-boxes. In the case of the transmission, the transducers that were utilized in Phase D verification were not all available during the Phase D known bad parts testing. In order to try to improve the data base for determination of the threshold level, Phase D data was re-analyzed using the new technique, and also using only those sensor outputs that were available during Phase D known bad parts flights. The net result was that if a threshold of 38 was set, the Phase D score improved from 3 out of 6 to 4 out of 6, and the Phase D known bad parts from 13 out of 19 to 16 out of 19 correctly identified. It is felt that if a larger data base for the known good transmissions were established, the score would further improve.

In summary, the following threshold levels were established, based on the re-analysis, for application to Phase E verification.

Engine	7	(based on using sensors 4, 7, and 8)
Transmission	38	(based on using sensors 125, 123, 126, and 129)
90° Gearbox	8	(based on using sensor 64 only)
42° Gearbox	10	(based on using sensors 59 and 61)

10.4.4.2 Phase E Verification Vibration Results

Applying the revised analysis technique as described in 10.4.1 to the Phase E verification data gave the following results:

Date	Run No.	Engine Weighted 4 σ	42° Weighted 4 σ	90° Weighted 4 σ	Transmission Weighted 4 σ
19 Nov.	205	5	1	0	7
24 Nov.	207	0	0	0	13
3 Dec.	209	0	15	67	40
7 Dec.	211	0	23	54	51
9 Dec.	213	0	21	42	19
10 Dec.	215	1	19	53	4
14 Dec.	217	0	4	18	8
16 Dec.	219	0	0	0	6

Based on the new threshold levels (reference 10.4-1) the following conclusions are drawn with regard to vibration faults:

Run No.	Engine	42°	90°	Transmission
205	Good	Good	Good	Good
207	Good	Good	Good	Good
209	Good	Bad	Bad	Bad
211	Good	Bad	Bad	Bad
213	Good	Bad	Bad	Good
215	Good	Bad	Bad	Good
217	Good	Good	Bad	Good
219	Good	Good	Good	Good

10.5 Transmission Test Cell - Tests to Destruction (Transmission Input Quill)

10.5.1 Test Conduct

During Phase E, a test program was conducted at the Bell Helicopter Company transmission test facility. The purpose of this test program was twofold:

1. To obtain vibration signatures on transmission input quill bearings that represented worse wear conditions than those tested at the ARADMAC test cells during Phase B,
2. To obtain vibration signatures on one or two degraded transmission input quill bearing assemblies as these bearings were run to destruction.

The magnetic tape data acquisition system, transducers, and transducer locations were essentailly identical to those used during the Phase B testing at the ARADMAC test cells.

Only the vibration data associated with the test to destruction of one input quill bearing assembly was analyzed by Hamilton Standard. The bearing assembly selected for testing had an incipient spalling failure judged to be typical of the characteristic failure mode observed in this type of bearing. After installation an initial vibration record was taken. Additional vibration records at the same RPM and power settings were then taken in five-hour increments until complete functional failure of the bearing was obtained.

The results of the testing to destruction of one input quill bearing assembly are shown in Figures 10-16, 10-17, and 10-18. These figures are power spectral density analyses of the vibration data and were generated

10.5.1 Continued

using a narrowband (2.5 hertz) constant bandwidth analog spectrum analyzer. Vibration data from transducer No. 125 an accelerometer mounted on the transmission input quill was used to generate the spectral analyses. The transmission operating speed and power conditions are indicated on the respective spectral analyses. Figure 10-16 illustrates the initial vibration spectrum obtained for zero operating hours; Figure 10-17 illustrates the vibration spectrum after the bearing assembly had been run for 20 hours; and Figure 10-18 illustrates the vibration spectrum after 40 operating hours. Functional failure of this bearing assembly occurred at approximately 42 operating hours.

10.5.2 Data Analysis

The spectrum is displayed in terms of PSD units (g^2/hertz) versus frequency (hertz). The PSD values are indicated on the left vertical axis of the spectral plot. Numbers in powers of 10 listed above some of the major spectral responses on these spectral analyses indicate a change of attenuation setting in a segment of the frequency spectrum. These changes in attenuator setting are necessitated to accommodate the wide dynamic range of the vibration signals.

As an example in interpreting these attenuator changes, reference is made to Figure 10-16. The full scale PSD value for the frequency range from 0 to 1000 Hz is indicated as 0.1 g^2/hertz. The value of the spectral peak at 715 hertz would be read from the chart as 0.058 g^2/hertz. The frequency segment from 1000 hertz to 2000 hertz indicates two major spectral

10.5.2 Continued

peaks at frequencies of 1268 hertz and 1338 hertz. The number 10 above each of these spectral peaks indicates that a change in spectrum analyzer attenuator setting was made and that the full scale PSD value for these peaks is 10 g^2/hertz. On this basis the peak value of the response at 1268 hertz is read as .18 g^2/hertz and the peak value of the response at 1338 hertz is read as 1.25 g^2/hertz. All three spectral analyses should be interpreted in a similar manner.

Table 10.1 is a listing of the significant bearing frequencies, then sidebands, the input quill gearmesh and its sidebands. A code was adapted to label the significant spectral responses on the graphs and is also indicated in Table 10.1. For example, B1 refers to the fundamental frequency associated with a pit on the bearing outer race, (F_o), B2 is the second harmonic of this frequency, ($2 F_o$), etc., +1B1 refers to the first upper sideband of F_o or $F_o + F_r$; -2B1 refers to the second lower sideband of F_o or $F_o - 2F_r$, etc. In a similar manner A1 refers to the fundamental frequency associated with a pit on the bearing inner race (F_i); and GM refers to the input quill gearmesh frequency. The frequency values listed in the table are more exact for Figures 10-17 and 10-18. These frequency values are somewhat low for Figure 10-16 since the operating speed for this spectral analysis is somewhat higher than that of Figure 10-17 and 10-18 as indicated.

Figure 10-16, the vibration spectrum for the input quill bearing assembly at zero operating hours (TO) has the major spectral responses labeled according to Table 10.1. The frequencies associated with a pit on the inner and outer bearing races are clearly defined. In addition

10.5.2 Continued

other spectral peaks associated with the upper and lower planetary gear meshes are also in evidence. These frequencies are transmitted through the transmission case and sensed by the input quill transducer as indicated previously in Section 6.5.3.6.

Figure 10-17 shows the vibration spectrum for the bearing after twenty operating hours (T20). The bearing pit frequencies (F_o and F_i) and their harmonics are still the dominant responses in the spectru. In addition sideband frequencies separated by the shaft rotation speed (F_r) are now more clearly evident. These sidebands are arrayed around the pit frequencies and their harmonics ($F_o \pm n\ F_r$, $2F_o \pm n\ F_r$, etc.).

Figure 10-18 shows the vibration spectrum of the bearing assembly after forty operating hours (T40). The sideband structure associated with the pit frequencies is now considerably more extensive and spectral peaks are located throughout the spectrum. Additionally, the level of vibration has risen sufficiently to require an attenuator change from 0.1 to 1.0. The extensive sideband structure has been labeled and reference to Table 10.1 allows the great majority of the spectral peaks to be identified. Reference to this table indicates that the frequencies associated with a pit on the inner and outer races, harmonics of these frequencies, and sidebands associated with these frequencies are the dominant responses in the spectrum.

Calculations of the bearing frequencies F_o and F_i at the operating speed shown in Figure 10-18 (5640 rpm) indicate that the basic repetion rate of the F_o and F_i frequencies should occur at 589 hertz and 820 hertz,

10.5.2 Continued

other spectral peaks associated with the upper and lower planetary gear meshes are also in evidence. These frequencies are transmitted through the transmission case and sensed by the input quill transducer as indicated previously in Section 6.5.3.6.

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Calculations of the bearing frequencies F_o and F_i at the operating speed shown in Figure 10-18 (5640 rpm) indicate that the basic repetion rate of the F_o and F_i frequencies should occur at 589 hertz and 820 hertz,

10.5.2 Continued

respectively. The spectral data indicates F_o occurs at 660 hertz and F_i occurs at 705 hertz. It is suspected that slippage of the rolling elements is occurring to account for this discrepancy. Any slippage of the bearing elements with respect to the races would cause F_p the train passage frequency to increase. An increase of F_p relative to F_r would have the effect of lowering F_i and increasing F_o proportionally. This is exactly the effect observed in the data, i.e., F_i decreases from 820 Hz to 705 Hz and F_o increases from 589 Hz to 660 Hz. Bearing contact angle variations, though they exist are not sufficiently large to account for such a large change in the basic frequency.

Frequency modulation is responsible for the generation of the extensive sideband structure indicated in Figure 10-18. The FM process has been treated in detail in Section 6.5.3.6. Relating the information in the three vibration spectra to mechanical events in the bearing is done as follows. Initially at TO the spectrum in Figure 10-16 is indicating that the bearing inner and outer races have a major fatigue pit or spall. These large pits one on the outer race and the other on the inner race cause a mechanical impact each time one of the rolling elements passes them. These impacts manifest themselves as the frequencies F_o, $2F_o$, etc., and F_i, $2F_i$, etc., in the vibration spectrum. The initial sideband structure is not very predominant because the FM effect is small. As indicated previously in Section 6.5.3.6 FM in a bearing would be caused by a general deterioration of the mechanical tolerances; the amount of the FM depending directly on the degree of deterioration.

10.5.2 Continued

In Figure 10-17 after twenty operating hours, the pits have become larger. This is indicated by a increase in the vibration level at the F_o and F_i frequencies. In addition this spectrum shows that the sideband structure attributable to the FM process is a bit more extensive than that of Figure 10-16. This fact indicates that the mechanical tolerances associated with the rolling elements have increased. This variation from ball center to ball center as the balls strike the pitted area causes the phase or frequency modulation process and the generation of additional sidebands separated from each other by the shaft rotational speed F_r (94 hertz in this case).

Finally, in Figure 10-18, the pitted area is larger still and mechanical deterioration of the bearing is setting in quite rapidly as evidenced by the general rise in the overall vibration level and the number of FM generated sidebands. Ball center to ball center variations are now much greater than in Figure 10-17. The spectrum shows that bearing failure is eminent and in fact functional failure of this bearing assembly occurred approximately two operating hours after this vibration signal was recorded.

Another interesting aspect to these vibration spectra is the reaction of the deteriorating bearing on the input quill gears and gearmesh frequency (F_{GM}). At the operating speeds for which these vibration signals were recorded, the input quill gearmesh frequency should occur at approximately 2730 hertz and should be the major response. However, even though there is a vibration component at 2730 Hz, it is not the major one. Most

10.5.2 Continued

of the vibration energy associated with the input quill gearmesh is located in the first and second upper sidebands (+161, +261, see Figure 10-18). This effect is observed in all three spectral plots. As Table 10.1 indicates the frequency geometry of the sideband structure, it shows that the gearmesh and its sidebands fall very close to that of the fourth harmonic of F_i and its sidebands. This makes it very difficult to separate the bearing frequencies of $4F_i$ and its sidebands from the gearmesh. However, the fact that the gearmesh sidebands are higher in value than the gearmesh frequency itself indicates that the FM effect is also present in this gearmesh. That is the bearing assembly because of its high wear and relatively loose mechanical tolerances is reacting upon the gear/shafting system causing the gearmesh frequency to be frequency modulated at the shaft rps (94 hertz). Not only then are the bearing frequencies indicators of malfunction but also the gearmesh frequencies.

Table 10.1 was generated by carrying the sideband structure associated with the various bearing and gear frequencies out to ± 10 sidebands. Close inspection of this table indicates that many sidebands associated with the bearing and gear frequencies are extremely close to one another making resolution of the separate frequencies difficult and in some cases impossible. For example, the sideband tabulation at the bottom of Table 10.1 lists the interfering sidebands associated with F_i and its harmonics.

10.5.2 Continued

The effect of this interference and its existence in a spectral analyses can be seen in the spectrum plot of Figure 10-18. Spectral peaks associated with the higher harmonics and sidebands of F_o and F_i are in many cases very irregular or jagged and wider than others. The spectral peaks associated with the fundamental frequencies of F_o and F_i (B1 and A1) and their sidebands are relatively interference free and as a result are narrower and monotonic.

10.5.3 Conclusions

The conclusions drawn from the vibration data on the input quill bearing tests to destruction are:

1) Vibration analysis would have flagged the bearing assembly as bad at least 40 hours before functional failure occurred.
2) Conclusive proof that the FM effect exists for both gears and bearings.